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POWER EXTENSION PACKAGE (PEP)
SYSTEM DEFINITION EXTENSION

Orbital Service Module Systems Analysis Study

VOLUME 2
PEP Preliminary Design Definition

AUGUST 1979   MDC G7870

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PREFACE

The extension phase of the Orbital Service Module (OSM) Systems Analysis Study was conducted to further identify Power Extension Package (PEP) system concepts which would increase the electrical power and mission duration capabilities of the Shuttle Orbiter. Use of solar array power to supplement the Orbiter's fuel cell/cryogenic system will double the power available to payloads and more than triple the allowable mission duration, thus greatly improving the Orbiter's capability to support the payload needs of sortie missions (those in which the payload remains in the Orbiter).

To establish the technical and programmatic basis for initiating hardware development, the PEP concept definition has been refined, and the performance capability and the mission utility of a reference design baseline have been examined in depth. Design requirements and support criteria specifications have been documented, and essential implementation plans have been prepared. Supporting trade studies and analyses have been completed.

The study report consists of 12 documents:

Volume 1   Executive Summary

Volume 2   PEP Preliminary Design Definition

Volume 3   PEP Analysis and Tradeoffs

Volume 4   PEP Functional Specification

Volume 5   PEP Environmental Specification

Volume 6   PEP Product Assurance

Volume 7   PEP Logistics and Training Plan Requirements

Volume 8   PEP Operations Support

Volume 9   PEP Design, Development, and Test Plans

Volume 10   PEP Project Plan

Volume 11   PEP Cost, Schedules, and Work Breakdown Structure Dictionary

Volume 12   PEP Data Item Descriptions

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ACKNOWLEDGMENT

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Section 1

INTRODUCTION

The extension phase of the Orbital Service Module (OSM) System Analysis Study was a 7-1/2 month study for continued development of engineering and programmatic data in support of the NASA–Johnson Space Center Power Extension Package (PEP) Project. The study centered on detailed PEP design definition to support Space Transportation System (STS) capability evaluation in a manner that is responsive to NASA missions and requirements.

The schedule and key milestones for the 10 study tasks are shown in Figure 1-1. Design Definition, Task 1, provided PEP system and subsystem preliminary design for the flight and ground equipment. The design reflected the results of Task 2, which consisted of 21 separate supporting analyses and trades.

Tasks 3, 4, and 5, Supporting Analyses, defined support requirements, provided operations analyses and trades, and assisted PEP supporting technology activi-
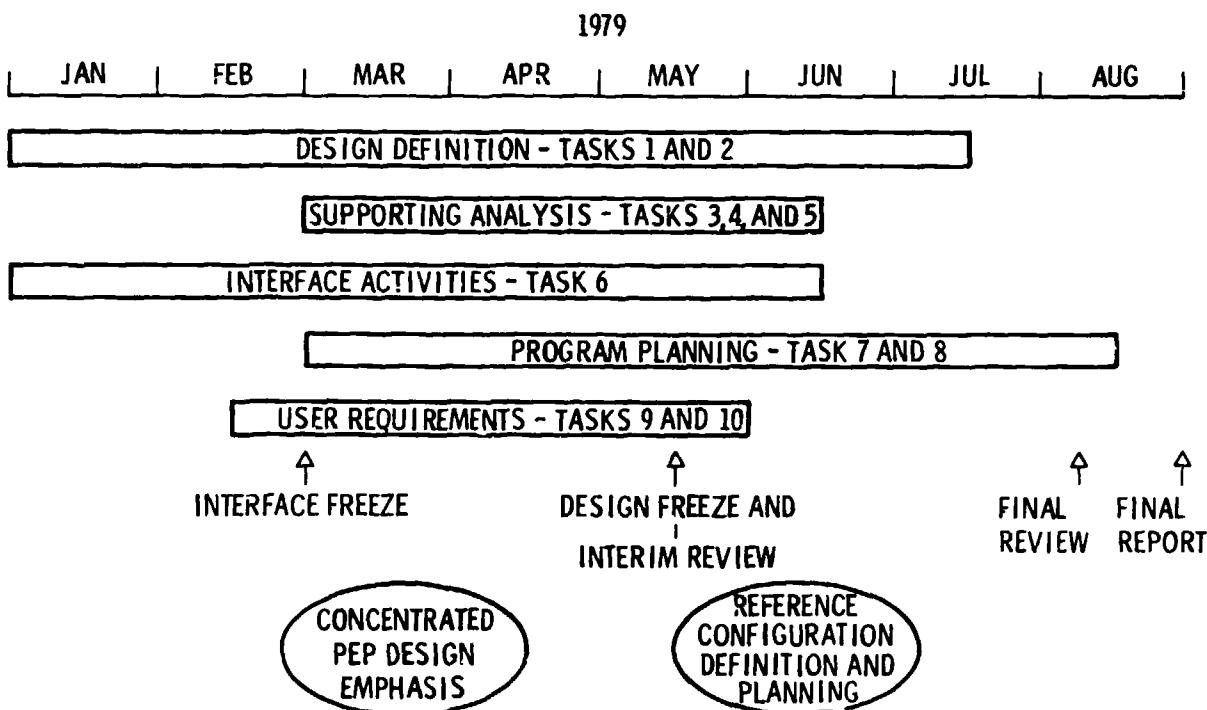
Figure 1-1. PEP Detail Definition Phase Schedule and Milestones for 1979

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ties. Outputs of these tasks included the Environmental Specification, Product Assurance Specification, Logistics Plan Requirements, Training Plan Requirements, and PEP operations support material.

PEP functional and physical interfaces with the remote manipulator system (RMS), the Orbiter, and the solar array were defined in Task 6, Interface Analysis.

Design, development, and testing were planned in Task 7, and the PEP Program Plan was updated in Task 8.

Task 9 updated performance requirements imposed upon PEP through involvement of potential users. A parallel objective evaluated the PEP design from a user standpoint. Task 10 included analysis to support the PEP design and the program comparison analysis that evaluated and compared PEP program options.

1.1 PEP MISSION REQUIREMENTS

PEP mission requirements are summarized in Table 1-1 and discussed in detail in Section 2. The requirements were derived during the original study for the Spacelab sortie missions scheduled in the October 1977 STS Mission Model. During the extension phase of the study, the requirements were expanded through

Table 1-1. PEP Mission Requirements

Item	Requirement
Mission	Shuttle sortie missions
Launch site	Integration and launch from Eastern Test Range (ETR) or Western Test Range (WTR)
IOC	1983
Power level	29 kW
Power type	28V regulated to Orbiter bus
Duration	20 days minimum at 21 kW and 55-deg inclination
Inclination	28.5 t⁻ 104 deg
Altitude	100 to 600 nm
Orientation	All attitude
Acceleration	No degradation of Orbiter low-g capability
RMS	No impact on payload use when PEP is not deployed

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the addition of other mission models and the results were reviewed with specific potential users.

The general requirement for PEP is to retain or increase current Orbiter capability while increasing available power and mission duration. General program guidelines include minimum impact to the Orbiter for scar and accommodations and retention of the RMS usefulness in its primary mode. Solar array concepts are to be based on NASA-developed solar electrical power (SEP) technology and commonality features are to be preserved for future related missions (i.e., SEP and Power Module).

1.2 REFERENCE PEP DESIGN

The reference PEP design is shown in Figure 1-2 in terms of major hardware elements. Section 3 discusses the PEP system and subsystem designs in detail.

The PEP is a solar electrical power generating system to be used on the Shuttle Orbiter to augment its power capability and to conserve fuel cell cryogenic supplies, thereby increasing power available for payloads and

POWER REGULATION AND CONTROL ASSEMBLY
- VOLTAGE REGULATORS/COLD PLATES
- SHUNT REGULATORS
- POWER DISTRIBUTION/CONTROL
- SUPPORT STRUCTURE

ARRAY DEPLOYMENT ASSEMBLY
- ARRAYS AND CONTAINERS
- MASTS/CANISTERS
- GIMBAL/SLIP RINGS/GRAPPLE
- SUN SENSOR AND CONTROLS
- INSTRUMENTATION
- CORE STRUCTURE

INTERFACE KIT
- RMS POWER CABLE
- ATTACHMENT FITTINGS
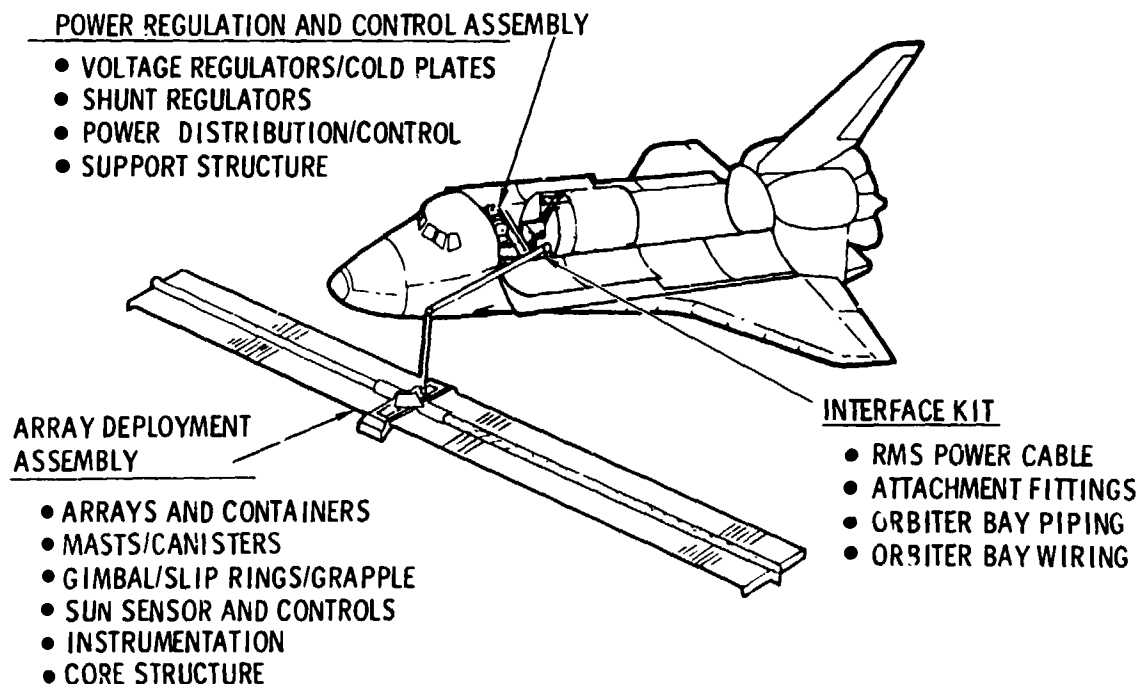- ORBITER BAY PIPING
- ORBITER BAY WIRING

Figure 1-2. PEP System

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allowing increased mission duration. This improvement can be seen in Figure 1-3, which gives capability for cryo tanks alone and with PEP. Mission durations of 20 days can be obtained for 7-kW power levels to the payload, 17 days for 15-kW payload power. Larger durations can be obtained for higher inclinations; for example, a mission of 48 days at 15-kw payload power level can be obtained when the payload is launched with a 97-deg inclination.

When required for a sortie mission, PEP is easily installed within the Orbiter cargo bay as a mission-dependent kit. When the operating orbit is reached, the PEP solar array package is deployed from the Orbiter by the RMS. The solar array is then extended and oriented toward the sun, which it tracks using an integral sun sensor/gimbal system. The power generated by the array is carried by cables on the RMS back into the cargo bay, where it is processed and distributed by PEP to the Orbiter load buses. After the mission is completed, the array is retracted and restowed within the Orbiter for earth return.

The PEP system, which consists of two major assemblies -- the array deployment assembly (ADA) and the power regulation and control assembly (PRCA) -- plus the necessary interface kit. It is nominally installed at the forward end of
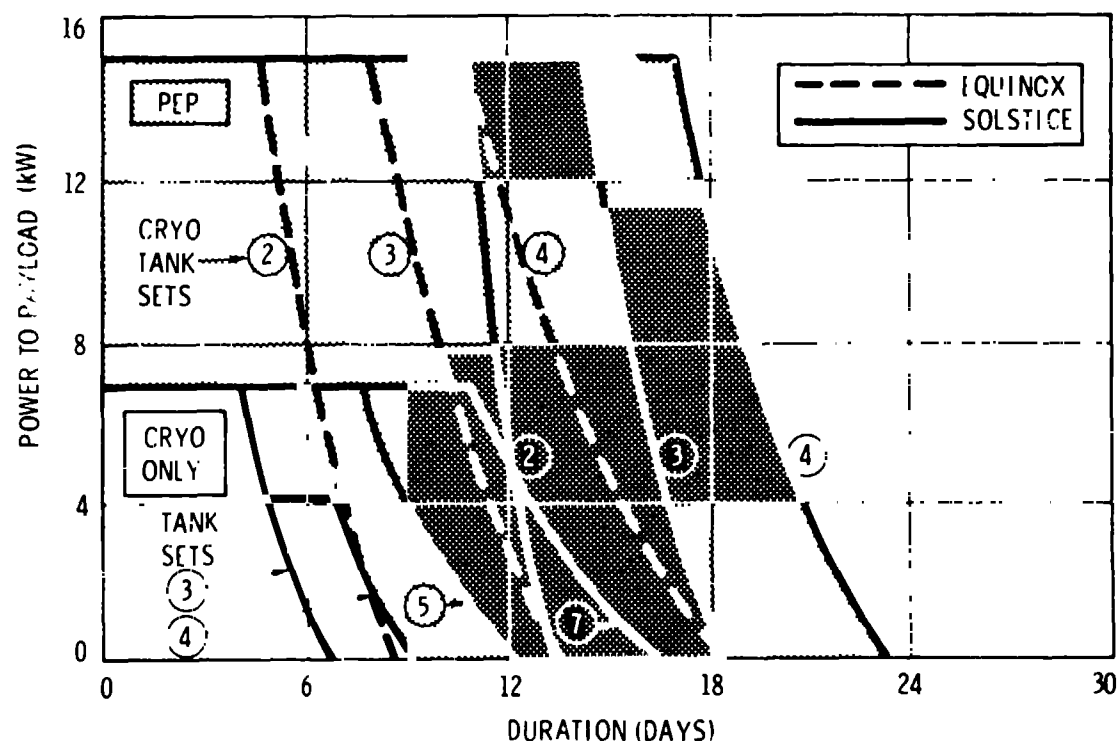
Figure 1-3. PEP Performance Benefits 55 Deg X 220 NM

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the Orbiter bay above the Spacelab tunnel, but can be located anywhere within the cargo bay if necessary. The ADA, which is deployed, consists of two light-weight, foldable solar array wings with their containment boxes and deployment masts, two diode assembly interconnect boxes, a sun tracker/control/ instrumentation assembly, a two-axis gimbal/slip ring assembly, and the RMS grapple fixture. All these items are mounted to a support structure that interfaces with the Orbiter. The PRCA, which remains in the Orbiter cargo bay, consists of six pulse-width-modulated voltage regulators mounted to three cold plates, three shunt regulators to protect the Orbiter buses from overvoltage, and a power distribution and control box, all mounted to a support beam that interfaces with the Orbiter.

PEP is compatible with all currently defined missions and payloads and imposes minimal weight and volume penalties on these missions. It can be installed and removed as needed at the launch site within the normal Orbiter turnaround cycle.

Section 2
MISSION REQUIREMENTS

The PEP mission requirements derived in the basic study have been reaffirmed
and further defined in the extension phase. User power, duration, and orbit
requirements have been the prime influencing factors on the PEP design and
were derived in the basic OSM study for the Spacelab sortie missions scheduled
in the October 1977 STS Mission Model. This mission model data was then
updated in the extension study. The orbit requirements were given directly in
the model. The power and duration requirements were derived by correlating a
time history of user needs and desires with the scheduled missions. These user
power and duration needs were derived from mission planning sources (i.e.,
5-year plans, Outlook for Space, recommendations from users, etc.). The power,
duration, and orbit requirements thus derived for PEP in the basic OSM study
are summarized in Figures 2-1, 2-2, and 2-3.
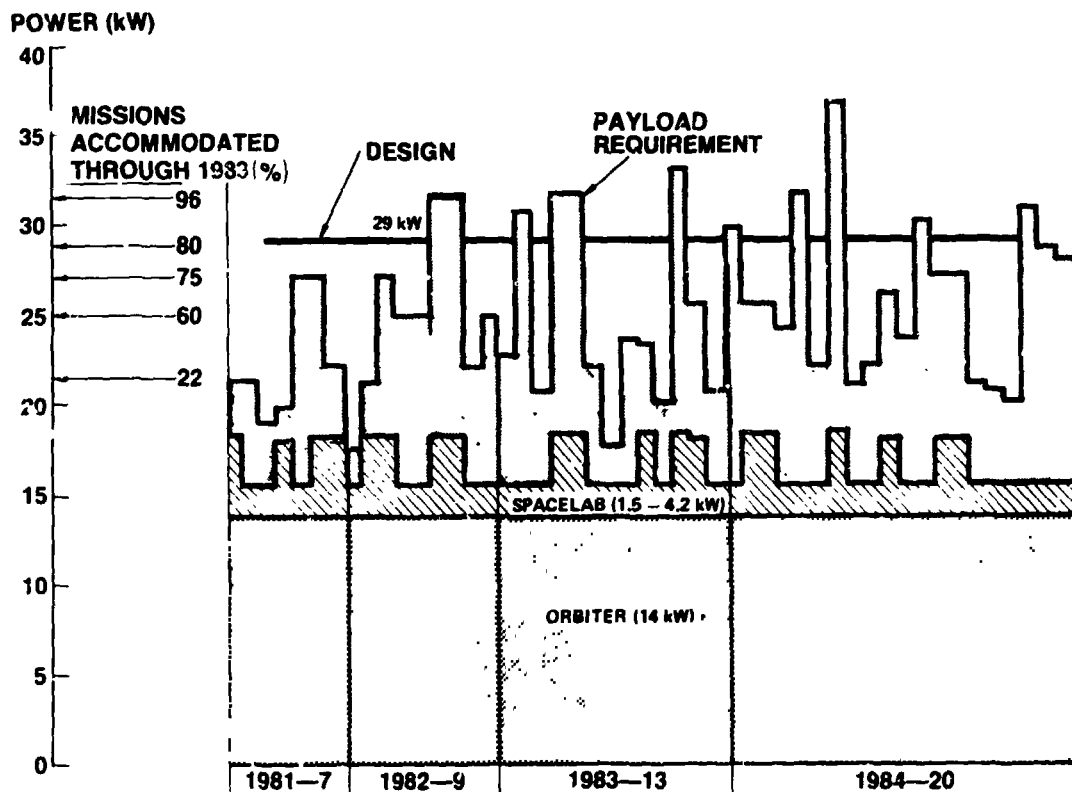

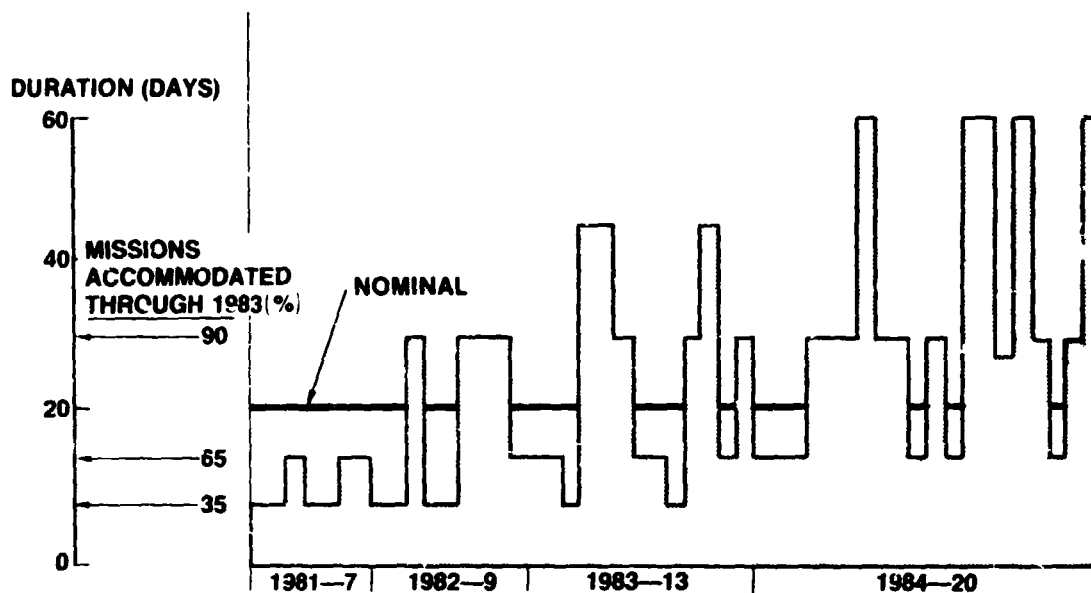Figure 2-1. Power Requirements—STS Mission Model (Spacelab Missions)

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DURATION (DAYS)

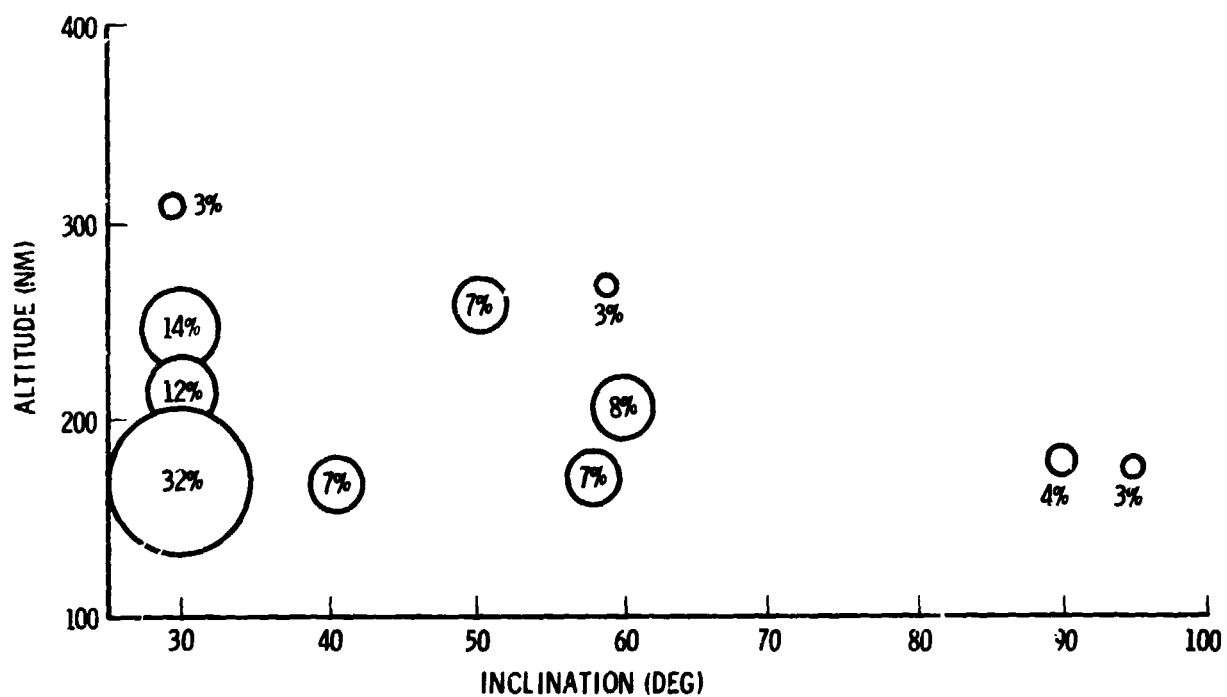Figure 2-2. Mission Duration—STS Mission Model (Spacelab Missions)

Figure 2-3. Altitude and Inclination Requirements

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The power requirements of Figure 2-1 consist of 14 kW for Orbiter housekeeping, an assessment of 1.5 to 4.2 kW for Spacelab support (i.e., pallet, igloo, module), plus the power requirements for the complement of payloads carried on each Spacelab mission. The power needs were obtained from knowledge of the payload requirement or by correlating the identified payload with projected user requirements obtained from the previously mentioned sources. As seen, the totals vary from 17 kW to 33 kW in the first 3 years. The suggested design range is overlaid on these requirements, capturing between 75% and 95% of the 1981-to-1983 missions. A 29-kW value accommodates 80% of the missions as defined, or 23 of the first 29, a figure that would appear to be a proper balance between increased capability offered and utilization over all the missions.

In Figure 2-2, the corresponding mission duration requirements are shown to vary from 5 to 45 days in the first 3 years. The design range that accommodates 65% to 90% of the missions is indicat i by the snaded area. A nominal requirement capability of 20 days duration was selected for design purposes although the PEP does have duration capability up to 48 days. The orbit altitude and inclination requirements from the October 1977 Mission Model are shown in Figure 2-3. The required inclination ranges from 28.5 deg to about 97 deg, with altitude from 150 nm to more that 300 nm. Additional benefits would be increased if inclination range was 28.5 to 104 deg and altitude range was increased 100 to 600 nm to be compatible with Orbiter.

During the extension phase of the PEP study, duration requirements were further defined and reviewed with specific potential users. They included various NASA center personnel cognizant of the payload needs of the first few scheduled Spacelab missions (SL-1 through SL-5), the NASA JSC Science Panel, and those responsible for mission planning in the Office of Space Sciences and Office of Space and Terrestrial Applications. A summary of the initial Spacelab Mission requirements is shown in Figure 2-4. These missions are currently scheduled to be accommodated by baseline Orbiter. The common note for each mission is that additional energy (power times duration) is needed to satisfy the basic mission needs or desired to more effectively utilize the planned experimental equipment. These data reaffirm the needs for increased power and duration determined in the basic PEP study.

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SPACELAB MISSION	PAYLOAD	ORBIT INCLINATION/ ALTITUDE	PAYLOAD POWER (kW)	DURATION (DAYS)	USER COMMENTS
1	LM AND PALLET	57 DEG/135 NM	6.7	7	ENERGY SHORT WITH 4 TANK SETS
2	3 PALLETS PHYSICS/ ASTRONOMY	57 DEG/225 NM	6	9	300 kWh SHORT WITH 5TH CRYO SET CURRENTLY OVERWEIGHT
3	LM AND PALLET PROCESSING, LIFE SCIENCE EART'. OBSERVATION	57 DEG/200 NM	7.7	8	ENERGY SHORT WITH 5 CRYO SETS
4	LM-LIFE SCIENCE PALLET-PHYSICS AND ASTRONOMY	28 DEG/160 NM	TBD	10	DESIRE INCREASED POWER, DURATION, WEIGHT
5	SM + 3 PALLETS PHYSICS AND ASTRONOMY	57 DEG/216 NM	TBD	7	WOULD LIKE MORE POWER AND DURATION

Figure 2-4. Early Spacelab Mission Requirements

Figure 2-5 shows a comparison of the mission model data sources available for use. The number of flights originally planned for the October 1977 Mission Model have been reduced by Jesse Moore data, the Flight Assignment Manifest (1300-0-6P), and other data. The distribution of flights, in terms of inclination and altitude regimes, remains unchanged. PEP must still accommodate the full Orbiter mission regime.

The above-mentioned contacts with potential users resulted in more detailed definition of how PEP would be used on particular missions. For example, the Spacelab 4 mission scheduled for 1982 is primarily a life science mission (long module) plus an aft-mounted pallet for physics and astronomy if that experiment could be accommodated. Though the specific payload for SL-4 is now being selected, candidate areas shown in Figure 2-6 illustrate the advantages of a capability for longer duration. Longer duration (up to 48 days) would allow a more complete set of data being gathered. The relative value of longer duration in terms of amount of data collected is shown in Figure 2-7. Some types of experimentation show no increase but most show a significant increase

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	MODEL	PARAMETERS	FISCAL YEAR											
			81	82	83	84	85	96	87	88	89	90	91	
	10-77	INCLINATION, ALTITUDE, SIZE, WEIGHT, DURATION	6	8	10	17	20	22	23	22	23	22	23	201
	POP 79-1	LAUNCH SITE	1	4	9	9	13	17	14	19	18	17	19	156
SORTIE	13000-0-6P	INCLINATION, ALTITUDE, CONFIGURATION	1	4	2									8
	J. MOORE		1½	4¼	4¼	6¼								17
	ORBITER OPTION 1		1	3	5									9

Figure 2-5. Mission Model Data Base

RESEARCH AREA	7-DAY MISSION	48-DAY MISSION
CARDIOVASCULAR DECONDITIONING	ADAPTIVE CHANGES STILL IN PROGRESS	STABILIZES IN LESS THAN 42 DAYS; ENTIRE MECHANISM CAN BE STUDIED
BONE DEMINERALIZATION	DETECTABLE ONLY. CHANGES STILL IN PROGRESS	MEASUREMENTS AT 14, 28, AND 42 DAYS PERMIT CORRELATION WITH OTHER LOSS FACTORS
LOSS OF RED BLOOD CELL MASS	CHANGES STILL IN PROGRESS	MAXIMUM IN 20-40 DAYS; CAN MEASURE DURING "TURNAROUND" PERIOD
GENETICS	LESS THAN ONE FRUIT FLY GENERATION	14 DAYS FOR EGG TO MATURE ADULT: 2-3 GENERATIONS
MORPHOLOGY AND DEVELOPMENT (E.G., FROG)	DEVELOPMENT STILL IN PROGRESS - 10%	80% DEVELOPMENT ADEQUATE FOR DEFINITE CONCLUSIONS ON GRAVITY EFFECTS

Figure 2-6. Life Sciences Duration Requirements of Spacelab 4

NOMINAL
10-DAY
FLIGHT

PROCESS VARIATION
WITH TIME

STEP CONSTANT
• SPECIMEN GENERATION
  STUDIES, I.E., FRUIT FLY

CONSTANT
• PLANT GROWTH
• ANIMAL DEVELOPMENT
• BONE AND MUSCLE
  DETERIORATION

DECREASING
• CARDIOVASCULAR
• RED CELL LOSS
• PERFORMANCE
• POSTURE/EXERCISE
• BIORHYTHMS

FINITE
• RODENT GESTATION

FIXED
• FLUID REDISTR'BUTION
• VESTIBULA'i CHANGES

0   10   20   30   40   50

DURATION (DAYS)

Figure 2-7. Number of Measurements and/or Amount of Data Increase With Time (Life Science Payloads)

in value as duration capability increases. Duration longer than a nominal 6 days with baseline Orbiter is of even more importance when one considers the experiment setup, zero-g accommodation (sickness), and take-down activities needed that would take about 1-1/2 days out of the nominal time period.

A review of the user requirements for orientation, stability, accelerations, and cleanliness indicate that the baseline Orbiter capabilities must be preserved. PEP therefore must not detract from these capabilities.

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Section 3
SYSTEM AND SUBSYSTEM DESCRIPTION

3.1 PEP SYSTEM

3.1.1 <u>System Requirements</u>
The system requirements to which the PEP concept is designed were derived from
the mission requirements discussed in Section 2 and summarized below. Detailed
requirements are documented in the systems specification.

Missions	PEP must be suitable for use as needed on Shuttle Sortie missions (i.e., Spacelab)
Launch Site	Capability of integration at and launch from ETR and WTR must be maintained
IOC	1983
Power Level	29 kW regulated 28 V to Orbiter bus
Duration	20 days at 21 kW at 55-deg inclination (up to 48 days capability is available)
Inclination	28.5 to 104 deg
Altitude	100 to 600 nm
Orientation	All attitude-pointing capability for payload bay-mounted sensors will be maintained
Acceleration	PEP will not degrade the low-g capability of Orbiter
RMS Impact	PEP will use the RMS yet not hamper its ability to sepately serve payloads as needed when PEP is not in use.

3.1.2 <u>System Performance</u>
The performance of PEP, in terms of power and duration capability, is a func-
tion of those parameters that influence the time in sunlight per orbit, i.e.,
inclination, altitude, launch, time of year, time of day, eccentricity, etc.
The actual mission performance is then dependent upon the specification of
these parameters. For illustrative purposes, the PEP power/duration capability
is discussed in the following paragraphs, as it is affected by these
parameters.

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The basic orbit inclination envelope of power/duration capability for PEP is shown in Figure 3.1-1. For an east launch (28.5-deg inclination), the duration varies from 10 to 19 days depending on the payload power used. At 7 kW (nominal level offered by baseline Orbiter), the duration capability is 12 days. This increases with orbit inclination to 20 days at 55 deg, 27 days at 80 deg, 42 days at 90 deg, and 48 days at 97 deg (sun-synchronous). These capabilities are for four cryo tank sets (the nominal Orbiter complement) and compares to the 5-to-6 day capability of the Orbiter without PEP using cryogens only for power. The capabilities given are for a 220-nm altitude and a 3-kW fuel cell idle level during the sunlight portion of the orbit. At 97 deg, the orbit is in full sun for the 48 days of mission duration; the user power can then be a full 15 kW to the payload, thus the vertical line on Figure 3.1-1. Duration is limited by the cryogen consumption in all cases; it is 48 days at sun-synchronous using the fuel cells at the idle level only.

The effect of orbit altitude on PEP capability is shown in Figure 3.1-2. The solid line indicates the mission duration capability as a function of altitude for several inclinations, assuming a 7-kW payload power load. The duration
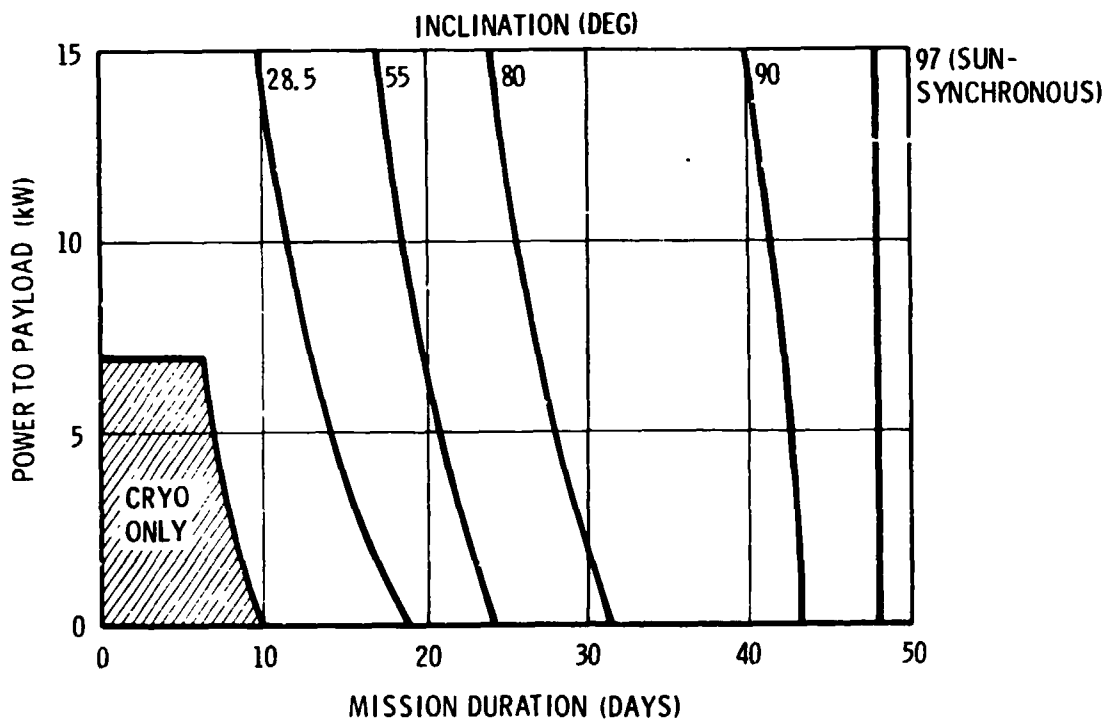

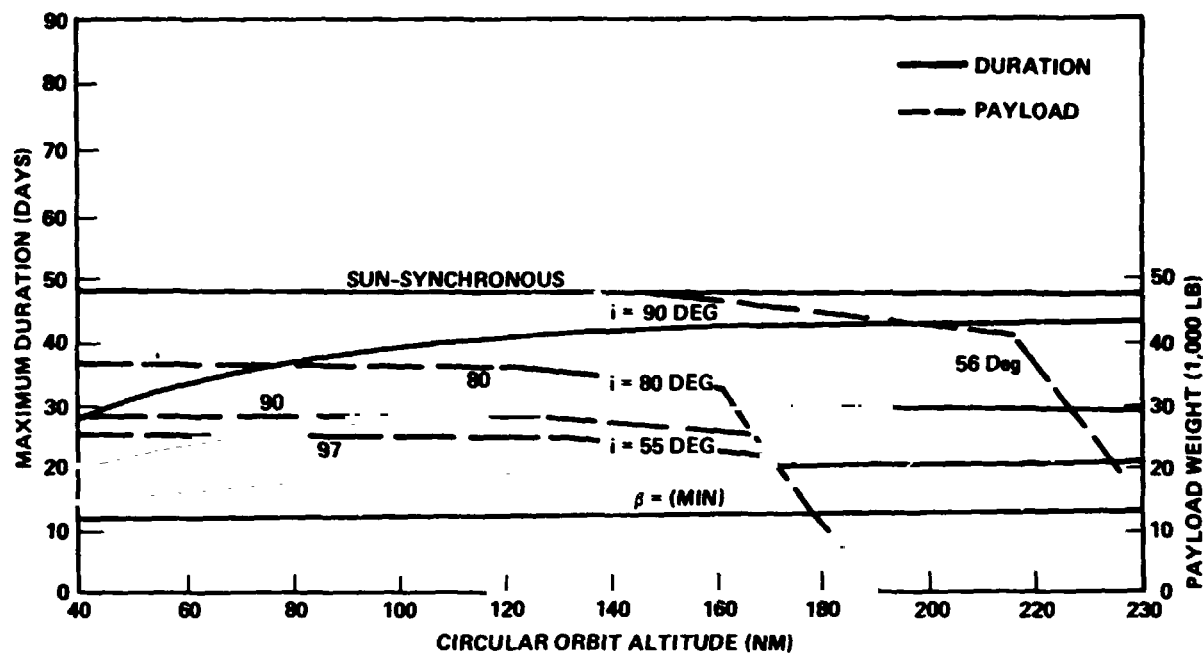Figure 3.1-1. PEP Detail Definition Phase Schedule and Milestones in 1979

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Figure 3.1-2. PEP Performance: Δh Effect 3-kW Fuel Cell Idle, 7-kW Payload Power, Four Cryo Sets

capability is not a strong function of altitude for the conditions shown. The dashed-line overlay is the projected Orbiter payload-to-orbit capability as a function of altitude for integral OMS capability. The 56-deg performance line is from ETR, the remaining from WTR. Orbiter payload capability begins to reduce markedly with altitude at the knee of the curve, where the integral OMS tank is full and altitude increase is then achieved by reducing payload. A plot of the duration-payload capability at these "knees" is shown in Figure 3.1-3 for missions launched from WTR. The lower line is for the projected Orbiter capability with a Block II external tank, some lightweight provisioning, etc. The upper line is for the JSC 07700, Vol. XIV capability. From the lower line, a 48-day, 97-deg mission at 170 nm can accommodate 20,000 lb; a 23-day, 70-deg mission at 155 nm can accommodate payloads up to 36,000 lb. For ETR launches, 40,000 lb can be accommodated on a 56-deg, 220-nm orbit with a 20-day duration.

The PEP capability for less than the nominal four cryo tank sets is shown in Figure 3.1-4. These data would be important for missions that require a maximum payload-to-orbit capability. Such capability would be achieved by configuring a high-payload Orbiter with, say, only two cryo tank sets. The

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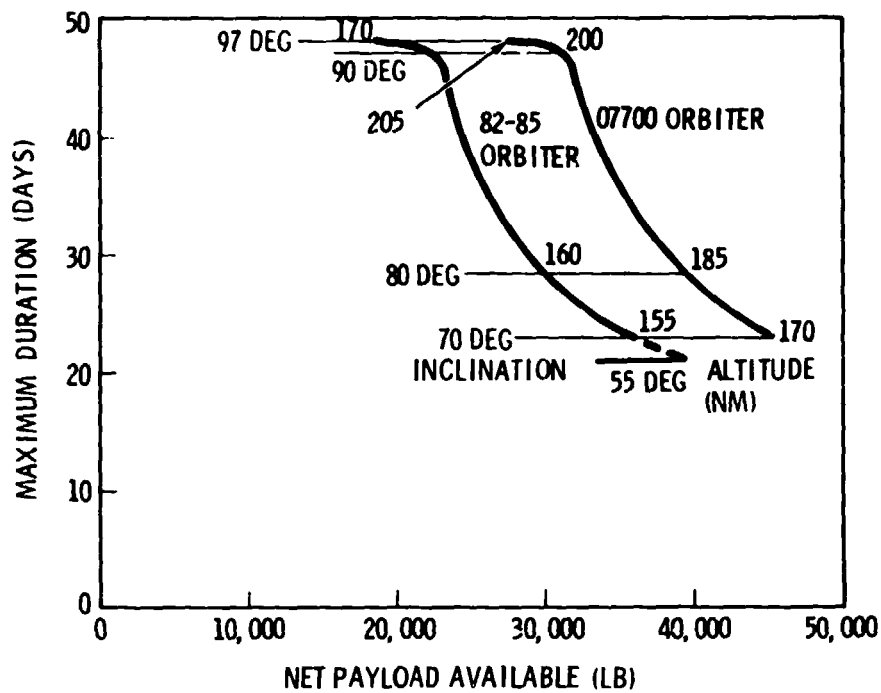
Figure 3.1-3. PEP Launch Capability from WTR
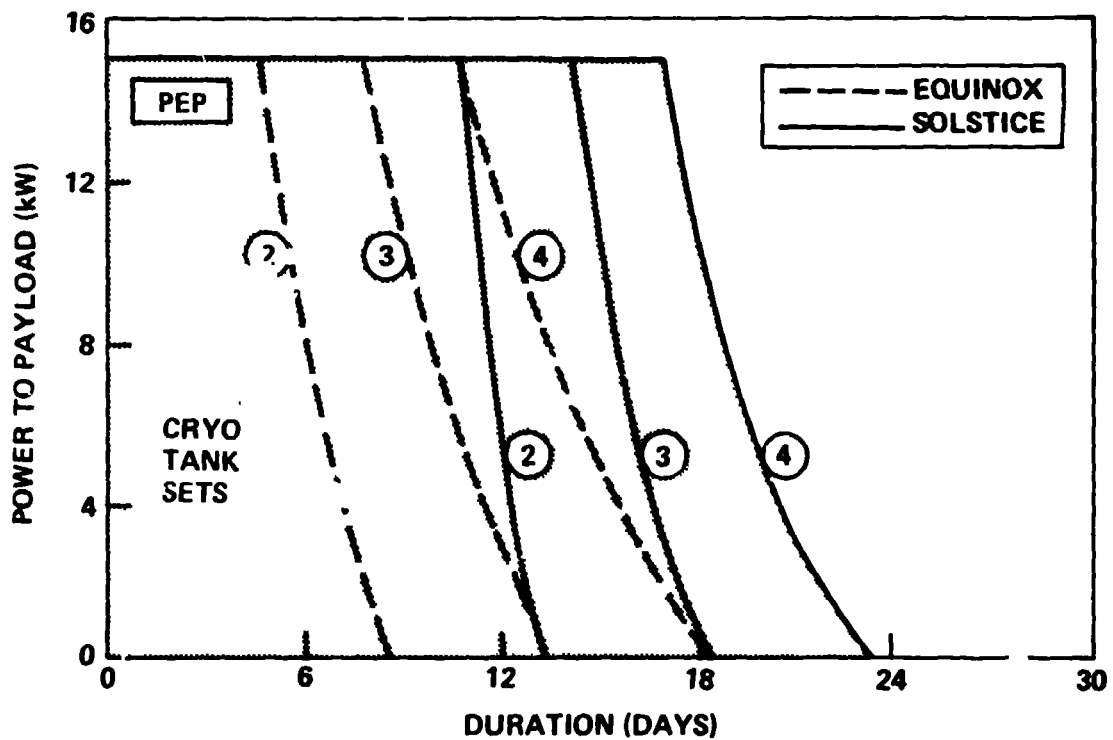
Figure 3.1-4. PEP Performance Benefits 55 Deg x 220 NM

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Orbiter would then be able to lift an additional 3,300 lb for a short delivery mission yet retain the option, with PEP, of flying extended duration to about 12 days at a 55-deg orbit. The mission could be extended to 20 days at 97 deg.

PEP performance has been quoted here as average power capability for a mission; however, the use of full power capability (29 kW with 15 kW for payloads) can be implemented while in the sun portion of the orbit without any duration penalty. On most missions, the time in the sun varies from about 63% to 100% of the time, so this full-power capability could be invoked much of the time. In addition, higher peak-power capability is available by operating the fuel cells with the solar panel on the daylight portion of the orbit. Over 50 kW could be supplied for short durations, up to 15 min. Payload wiring and cooling provisions to handle the increased loads would, be added in kit form.

The duration capability is affected by time of year because of available sun angle relationships. Figure 3.1-5 shows the duration achievable with PEP as a function of time of year for various orbit inclinations for a 220-nm altitude. A favorable sun-synchronous condition can be achieved any day of the year.
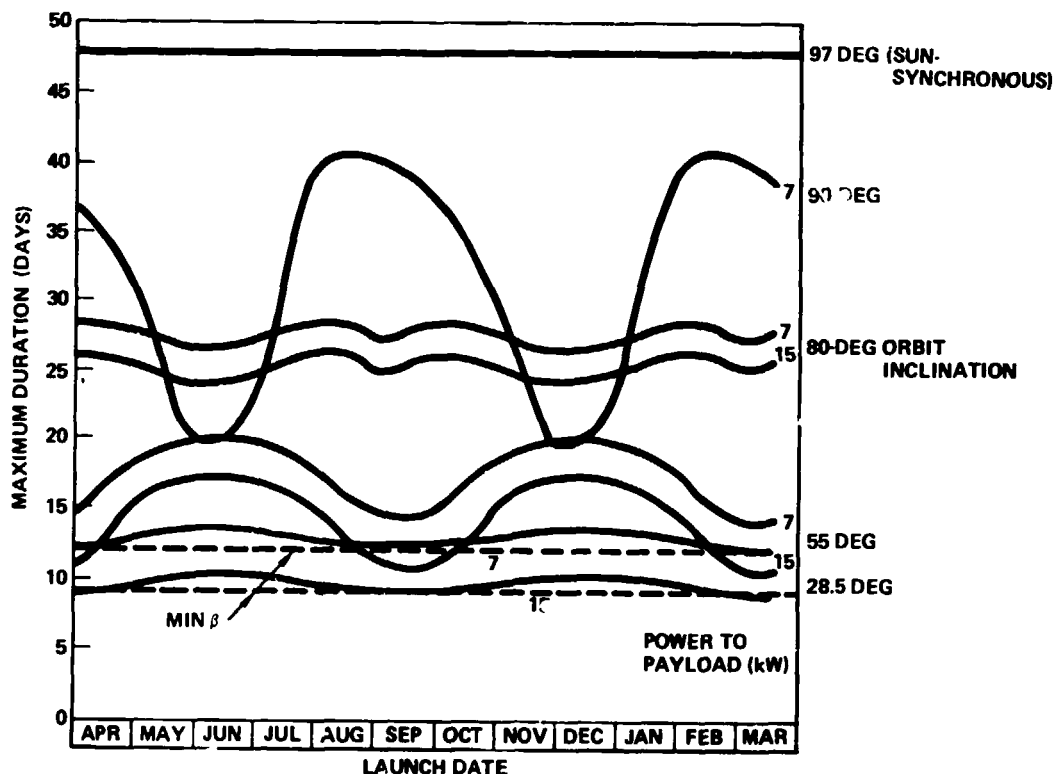
Figure 3.1-5. PEP Performance — Launch Date Effects Attitude = 220 NM

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Solstice conditions are most favorable for 28.5-deg and 55-deg orbits because they allow the achievement of high (near 90 deg) Beta angles. Beta angle is the angle between the sun line and its projection on the orbit plane. At 80 deg, the solstice and equinox conditions are similar enough to give a fairly flat variation with time of year. At 90 deg, the equinox portion is most favorable. These capabilities are contingent upon launching at the proper time of day to obtain favorable Beta angle relationships.

The effect of launch time of day on duration is shown in Figure 3.1-6. Each curve is shown for the most favorable time of year for each inclination, i.e., solstice or equinox. At sun-synchronous (~97 deg), the optimum time of launch is near 0600 hr. The window is fairly long, ~2 hr, and it increases as the desired duration capability is reduced. For missions of lower inclination, the launch window increases. These windows compare favorably with a normal launch window of about 10 min needed to effect rendezvous at high inclination. Figure 3.1-7 shows the effect of time of day of launch and duration capability for a 97-deg 220-nm orbit. An equinox condition is shown to illustrate the earth surface sun illumination angle relationships. An 0600 launch would place the orbit at the terminator and allow the longest mission duration. For lower sun
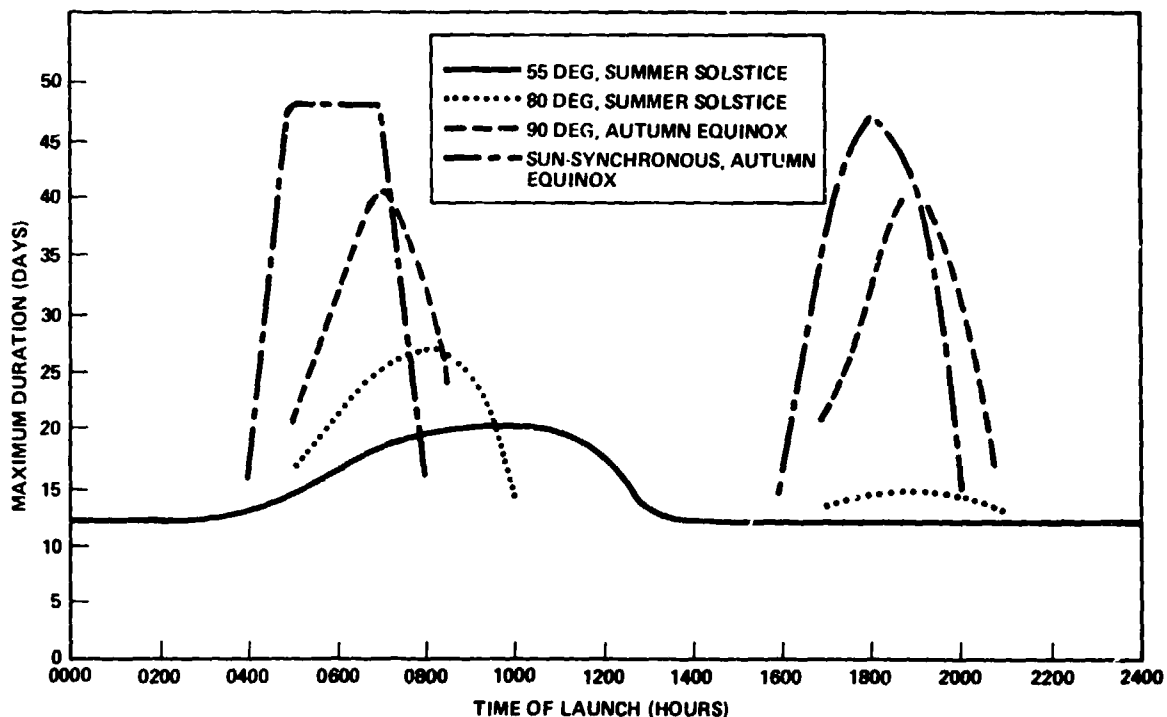
Figure 3.1-6. Mission Duration — Launch Window Effect 3-kW Fuel Cell Idle, 4 Cryo Sets; h = 220 NM, 7 kW to Payload

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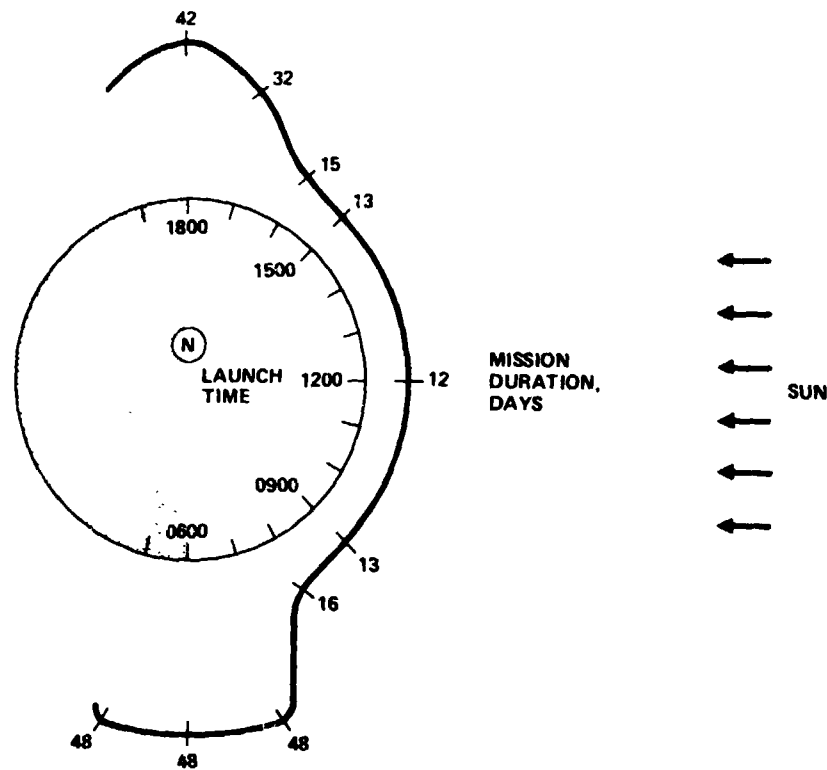
Figure 3.1-7. Mission Duration — Sun Illumination 97-Deg, 220-NM Orbit, 7-kW Payload Power, Autumnal Equinox

angles, the duration capability decreases to 12 days for a noon launch. Evening launches then allow an increase up to 42 days at 1800 hr. Thus the effect of particular viewing needs on mission duration can be determined.

The effect of elliptic orbits on duration capability was examined for PEP. Figure 3.1-8 illustrates the sensitivity for a 55-deg orbit. At a 100-nm circular orbit, the duration capability is 17 days. If apogee is increased, while maintaining a 100-nm perigee, the duration capability is increased as shown (i.e., 1 day for each 100-nm increase in apogee). The Orbiter limits achievable apogee altitudes to about 500 nm for integral orbital maneuvering system capability; thus, the duration increase is limited to about 5 days over the nominal (reference 17 days). Other Orbiter mission performance capabilities, such as acceleration, pointing, maneuvering, and payload heat rejection, have been examined with respect to PEP. In all cases, the nominal Orbiter capability has been maintained or improved by the use of PEP. Figure 3.1-9 shows an acceleration capability comparison for a low-g mission where the Orbiter is flying an X-axis along the local vertical (gravity-gradient mode). At 160 nm,

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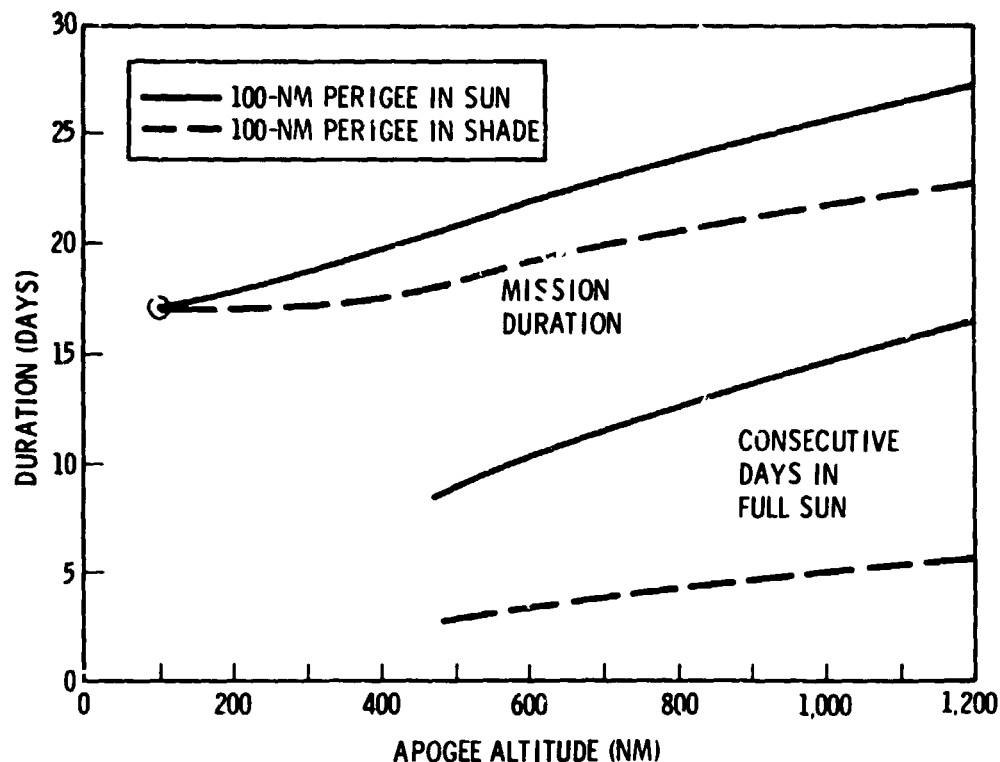
Figure 3.1-8. Elliptic Orbit Capability, i = 55 Deg

| | ACCELERATION (G'S) | |
DISTURBANCE	ORBITER ONLY	ORBITER WITH PEP
AERODYNAMIC (~160 NM)	5.5×10^{-7}	1.2×10^{-6}
PEP GIMBAL DRIVE TORQUE		1.8×10^{-6}
GRAVITY-GRADIENT AND AERO-FREE OSCILLATIONS	2×10^{-6}	2×10^{-6}
CREW MOTION		
17 LB PUSH-OFF	10^{-4}	10^{-4}
2 LB PUSH-OFF	10^{-5}	10^{-5}
VERNIER REACTION CONTROL SYSTEM	10^{-4} TO 10^{-3}	10^{-4} TO 10^{-3}
PRIMARY REACTION CONTROL SYSTEM	4×10^{-3} TO 4×10^{-2}	4×10^{-3} TO 4×10^{-2}

Figure 3.1-9. Acceleration Levels

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the drag acceleration with PEP is about twice that of Orbiter alone, but the level is still below that seen by payloads due to expected gravity-gradient or aero induced oscillations.

Attitude control and pointing capability is unaffected by PEP in terms of accuracy. Analysis indicates that the vehicle may reside toward one side of an attitude dead zone a bit more with PEP, yet would still be within the nominal accuracy requirement. A negligible effect on propellant budgeting is needed, due to PEP, for expected operating altitudes (>100 nm). The payload heat-rejection capability, using PEP, is increased over the nominal Orbiter because the fuel cells operate at idle level when the orbit is in the sun; thus, the amount of waste heat from the fuel cell that must be dumped is reduced. This capability gained can then be used by payloads as needed.

3.1.3 System Description

Figure 3.1.3-1 illustrates the PEP concept in a representative configuration. The major elements of the system are the ADA, the PRCA, and an interface kit which contains PEP-to-Orbiter and RMS interface provisions, including RMS power cable, attachment fittings, and Orbiter bay wiring and plumbing. PEP system characteristics are summarized in Table 3.1.3-1.

ROTATING
CANISTERS

SPACE
LAB

(TUNNEL
OMITTED
FOR CLARITY)

FORWARD ⊙

ARRAY DEPLOYMENT
ASSEMBLY
• STRONG BACK
  CONFIGURATION

POWER REGULATION AND
CONTROL ASSEMBLY
• BEAM MOUNTED
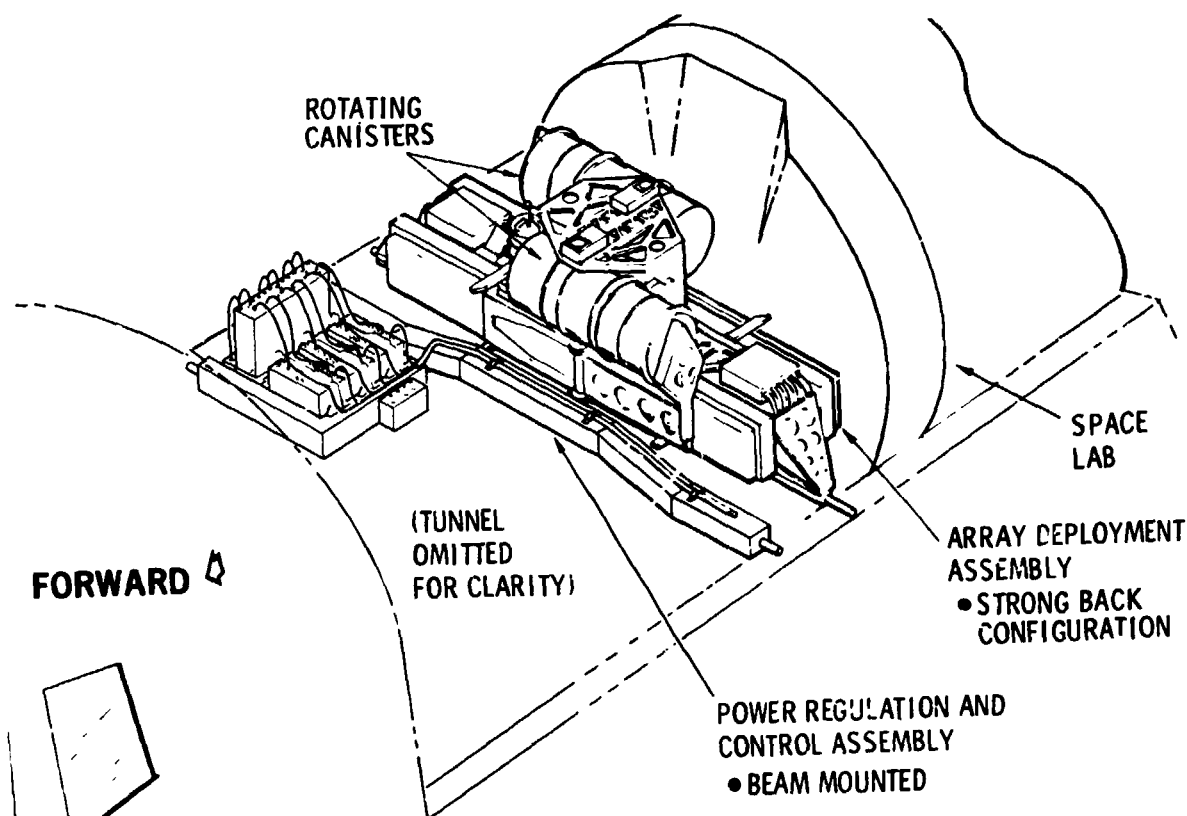
Figure 3.1.3-1. PEP Reference Installation (Two-Beam Configuration)

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Table 3.1.3-1. PEP System Characteristics

Power and duration	21 kW, 20 days at 55-deg inclination
(PEP/Orbiter)	29 kW, 17 days at 55-deg inclination
	29 kW, 48 days at 97-deg inclination
PEP power output	
Load interface	2(.3 kW gross BOL.
PEP interface	26.9 kW gross BOL
Output voltage	32.6 V +0.1 (PEP regulation mode)
Array size	2 SEP-type wings, 3.84 X 37.8 meters each
Stowage location	Over Spacelab short or long tunnel, aft location optional
Stowage configuration	Dual wingbox; rotating canister; trunnion/latch attachment
Deployment	Remote manipulator system
Array rotation	Separation gimbal/torquer drive using sun sensor control RMS inactive except during Orbiter maneuvers
Weight	PEP, 2,266 lb; Orbiter attach fittings, 85 lb
Heat rejection	Uses Orbiter fluid loop and radiation. Flash evaporator supplement; some orientations

Both the ADA and the PRCA are normally stowed at the forward end of the Orbiter bay, but can be located anywhere within the cargo bay if needed. The forward location is the most crucial design point for PEP because of the proximity of Spacelab module and tunnel or pallets and igloo for the all-pallet configuration. The Orbiter/PEP retention provisions are shown in Figure 3.1.3-2.

The ADA shares two standard Orbiter bridge fittings with the Spacelab short tunnel and one standard bridge fitting with the Spacelab module; three remotely operated lightweight custom retention latches retain the ADA within the Orbiter and allow the ADA to be mounted over Spacelab standard pallets. The PRCA is installed forward of the ADA on two custom lightweight bridge fittings which provide clearance for mounting adjacent to the RMS mounting; the PRCA trunnions are locked into bridge fitting journals for lateral load reaction.

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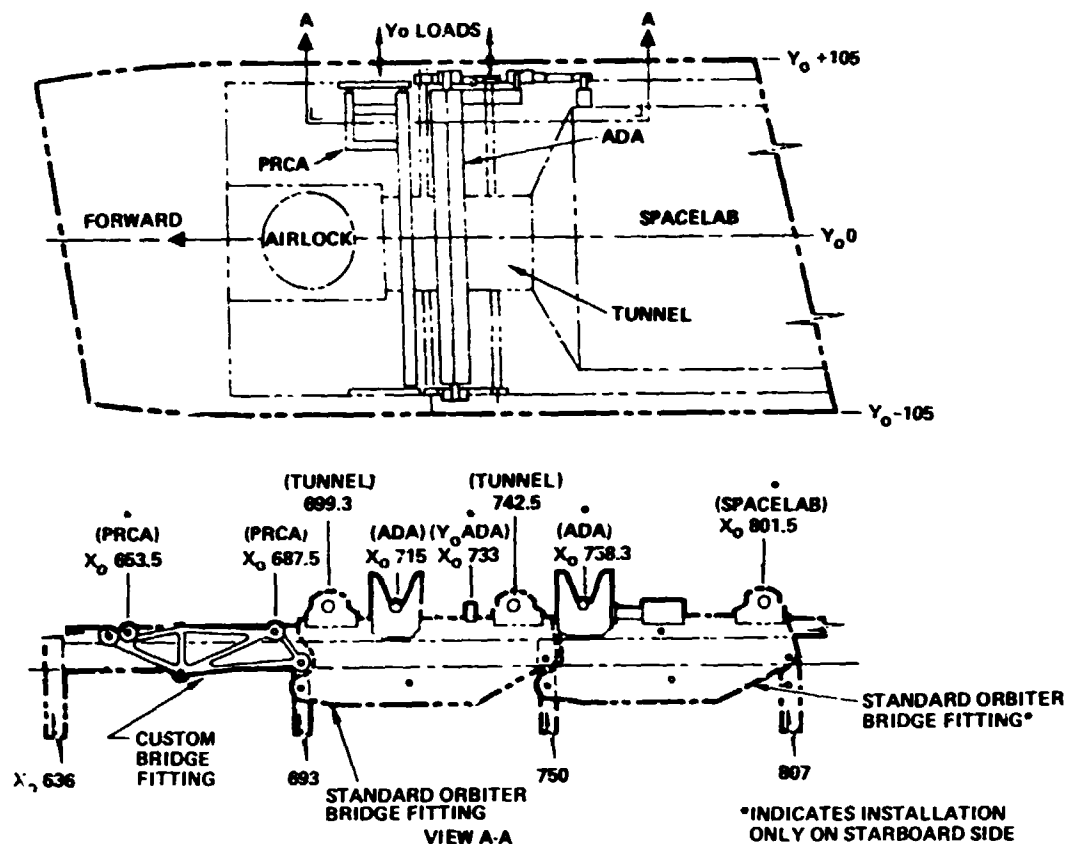
Figure 3.1.3-2. PEP Retention Provisions

On orbit, the ADA 's grasped by the ᴧᴹS using a standard end effector and moved outside the Orbiter bay. The ADA is then positioned to allow the mission and payload orientation requirements to be met while allowing solar array alignment normal to the sun line. The selected location could be at the left or right side of the Orbiter, below, or in front as needed, to best satisfy mission objectives. When the optimum position is established by the RMS, the RMS joint brakes are maintained locked throughout the selected orientation sequence, e.g., Y-axis perpendicular to the orbit plane. The deployed solar array is then aligned normal to the sun line by the two PEP gimbals. The outer Alpha gimbal is placed by the RMS wit its axis perpendicular to the orbit plane; this then allows Alpha gimbal 360-deg drive to rotate as needed in the plane of the orbit. The 0 to 90 deg Beta gimbal allows the array to adjust toward the sun, accounting for orbit Beta angle variations. Power is transferred from the array across slip rings on the Alpha gimbal, through the remotely activated RMS end effector power umbilical, and along power cables attached to the RMS to the RMS shoulder, from where it is fed into the PRCA. There the power is controlled, regulated, and distributed to the Orbiter buses

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for use by the Orbiter and its payloads. The design enables the RMS to retain
the ability to serve payloads when PEP is not operational.

An exploded view of the ADA design is shown in Figure 3.1.3-3. The core struc-
ture is a box beam that provides for transverse attachment across the Orbiter
payload bay. Two solar array wing assemblies are attached to opposite sides of
the beam. The deployment canisters are mounted on top of the beam and will
undergo a 90-deg rotation prior to deployment. Also mounted to the core struc-
ture are the two diode assembly packages, which provide array module isolation
and interconnect; the two-axis gimbal/slip ring/RMS grapple fixture, which
provides array orientation, power transmission and RMS attachment; the sun
sensor and sun sensor processor, which derive control signals for array posi-
tioning; and the pointing and control electronics, which drives the gimbal and
provides the signal processing to feed the Orbiter displays and controls.

In performing this phase of the PEP study, several ADA configuration
alternatives have been examined. Figure 3.1.3-4 illustrates four generic
types. All of these options are feasible; they differ mainly in the method of
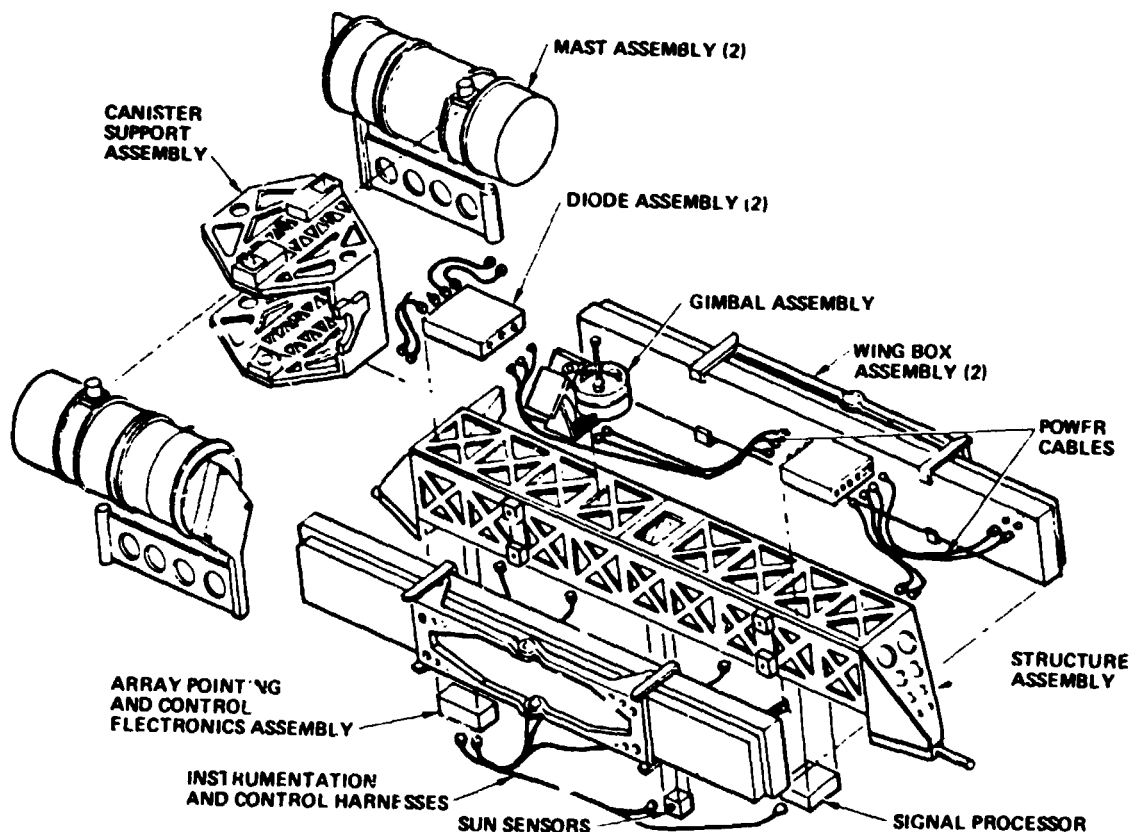
Figure 3.1.3-3. PEP ADA — Reference Configuration

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SELECTED REFERENCE CONFIGURATION

SUBASSEMBLY CONCEPT FOR STRONGBACK CONFIGURATION

SUBASSEMBLY CONCEPT FOR SINGLE BEAM CONFIGURATION

SUBASSEMBLY CONCEPT FOR INTEGRAL WING BOX CONFIGURATION
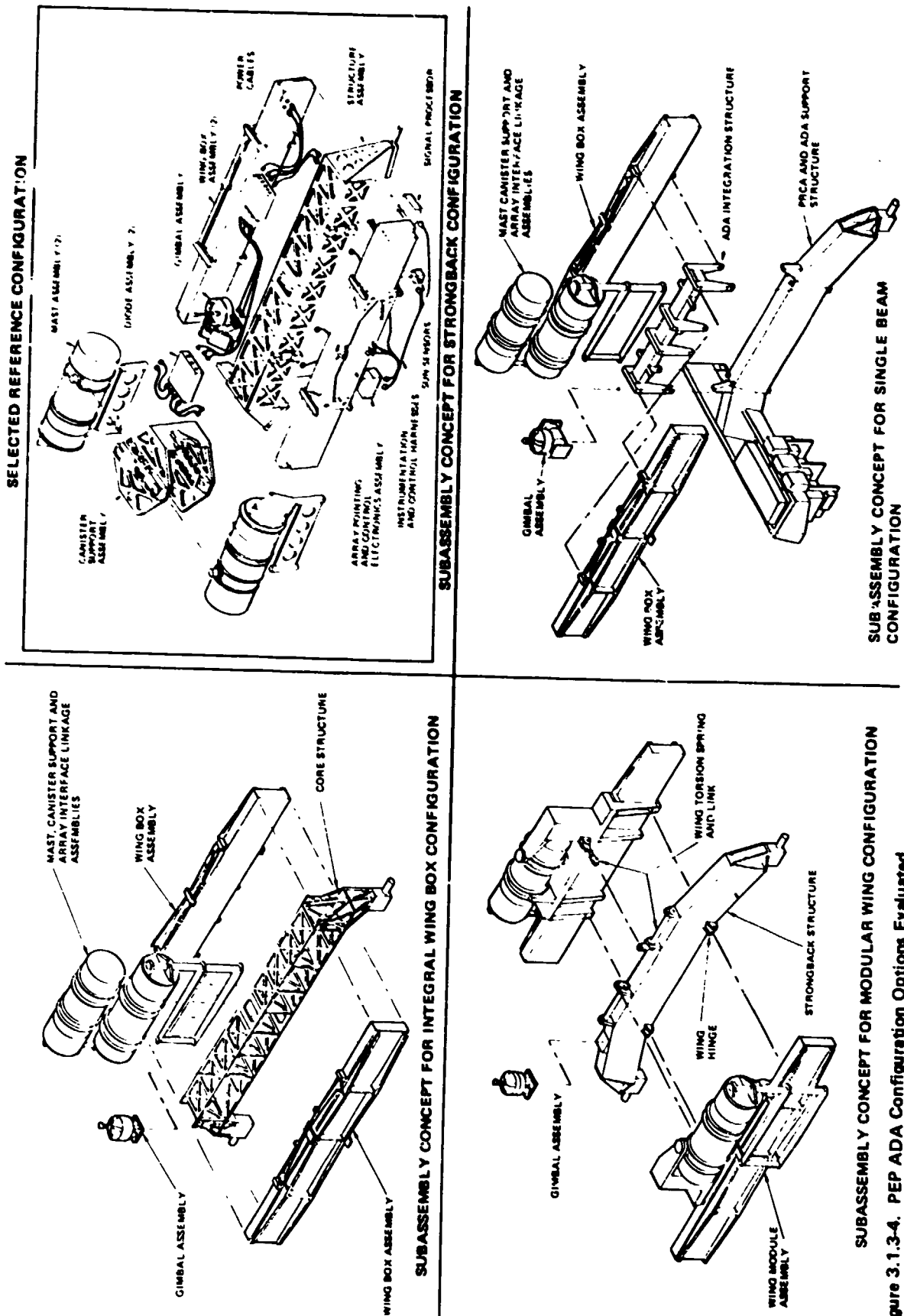
SUBASSEMBLY CONCEPT FOR MODULAR WING CONFIGURATION

Figure 3.1.3-4. PEP ADA Configuration Options Evaluated

25

mounting canister and wing box assemblies to the core structure. The configuration in the lower left side of the figure is a modular concept that allows each wing box and pair of canisters to be assembled as a module and integrated before attaching to the strongback structure. A similar configuration has been proposed by the Lockheed Missiles and Space Company. The strongback configuration, upper right, represents the type of configuration selected as a reference baseline for the study. This lightweight concept incorporates a central support mechanism for the canister/mast assemblies. This mechanism also houses the canister rotation linkages and the compliance springs for controlling the system's structural response.

Any of the ADA configurations shown can be outfitted with fixed (off-center) or rotating canister assemblies.

The solar array wing shown in Figure 3.1.3-5 is based on the NASA SEP array concept; it consists of 50 hinged panels per wing of 2X4 cm solar cells attached to a flexible substrate. Although the current baseline assumes a wraparound cell configuration, both conventional cells and larger cell sizes

DESIGN DATA

● Si CELL	– 8 MILS THICK
	– 2 X 4 cm WRAPAROUND
	– 12.8% EFFICIENCY
● COVER	– 6-MIL MICROSHEET
● SUBSTRATE	– 1/2-MIL KAPTON WITH INTERNAL WIRING AND WELDED INTERCONNECTS
● WING SIZE/AREA	– 3.84 X 37.8 METERS/ 145 METERS^2
	– 50 PANELS PER WING
● OUTPUT OF WING	– 16.4-kWe RAW POWER
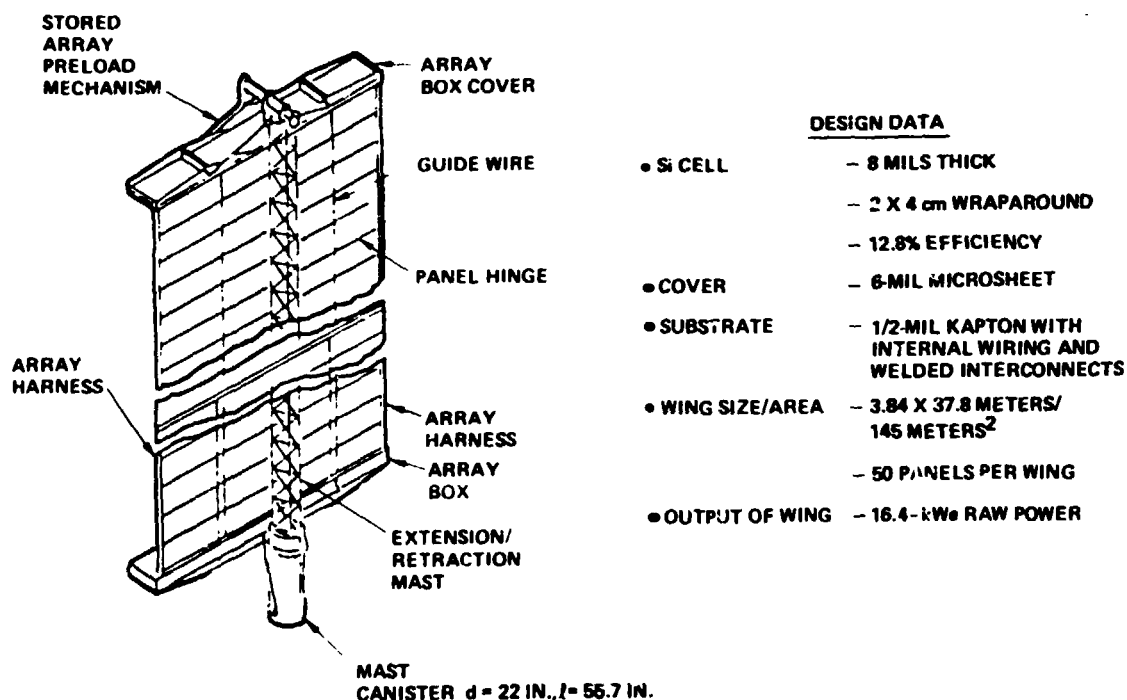
Figure 3.1.3-5. PEP Array Characteristics Current Baseline

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have also been investigated. Final design selection will occur during early Phase C/D.

he array is deployed and retracted by actuation of the deployable mast shown in Figure 3.1.3-6; the mast consists of a composite triangular truss, stored helically wound in a canister. During mast extension or retraction, the array folding is controlled by guide wires; when fully extended, the 3.84X36-meter array wing is kept under tension by the mast to assure the required flatness.

The mast is deployed from its canister by redundant motors driving through a gear box that provides for two-speed operation. During the first 2 ft of mast extersion, the canisters are unlatched and auto-rotated into deployment position at slow speed; after latching in this position, the array is fully extended at high speed. This sequence is reversed during retraction. During contingercy operation, array retraction and extension can be performed manually by an EVA crewman.

The mast canister assembly is attached to the support beam through a compliant mount which controls the frequency response of the deployec array to Orbiter-
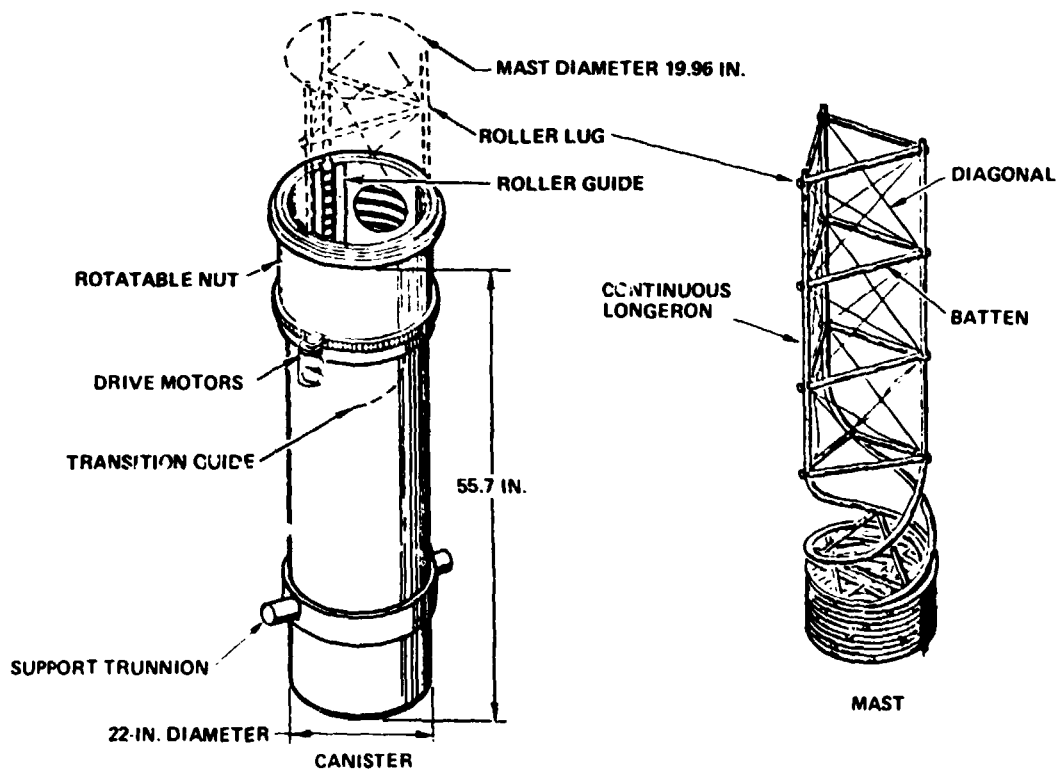
Figure 3.1.3-6. Mast/Canister Assembly

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induced loads in two axes (in the array plane and perpendicular to the array plane). The compliant mount is locked out when the array is not deployed.

When PEP is aboard the Orbiter, the Orbiter digital autopilot (DAP) software inhibits the length and/or frequency of vernier reaction control system (VRCS) pulses to control both available Orbiter rates and VRCS plume loads on the array. In conjunction with the compliant mount, this restriction assures that the array/mast will not be subjected to excessive loads and that the RMS brake torques will not be exceeded.

The gimbal/slip ring/grapple assembly and its relationship to the RMS system are shown in Figure 3.1.3-7. The standard RMS end effector mates physically with the grapple fixture on this assembly and a special remotely actuated umbilical mates with the special PEP harness installed along the RMS; the RMS modifications do not interfere with normal RMS operations when PEP is not in use.
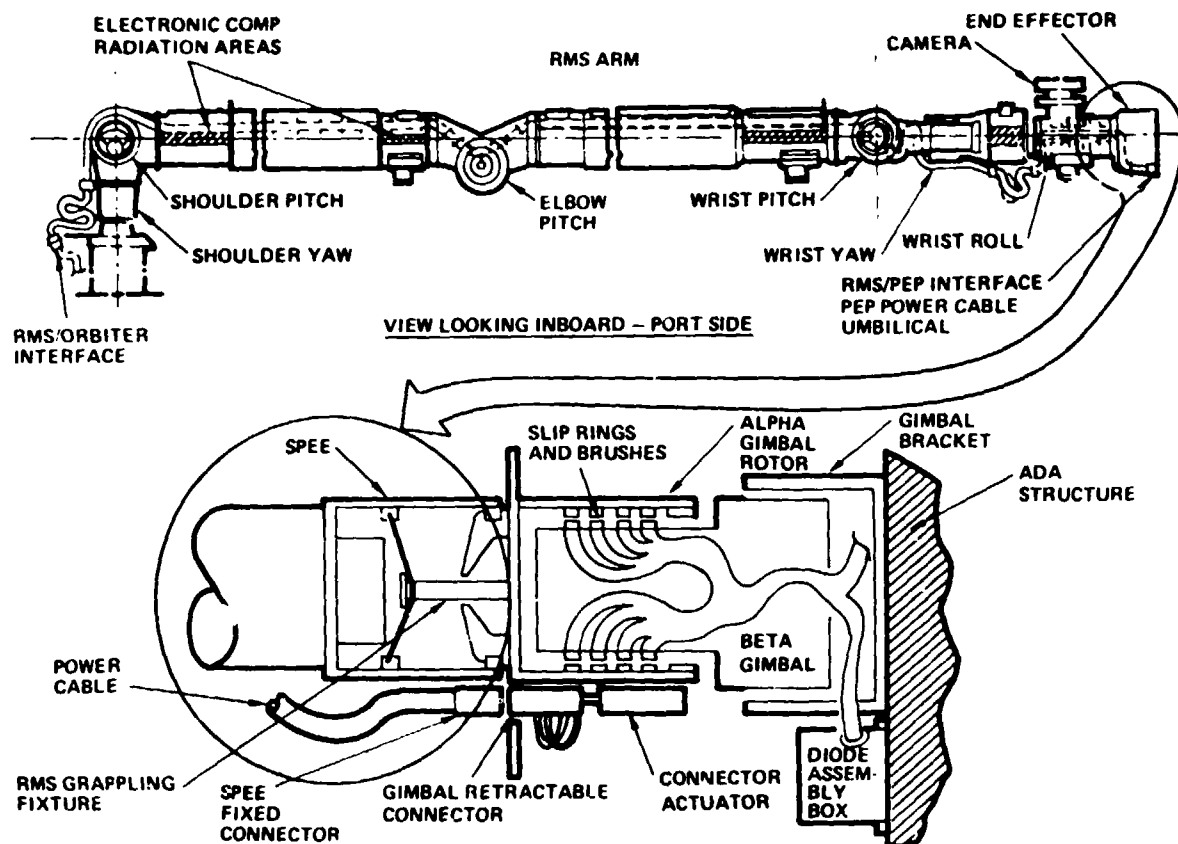
Figure 3.1.3-7. Gimbal/Slipring/Grapple Assembly

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All safety-critical latches and actuators that are incorporated in the ADA are capable of manual EVA operation in an emergency.

The avionics equipment on the support beam (sun sensor, sun and signal sensor processor, and pointing and control electronics assembly) provide the system control interface with the Orbiter through the multiplexer/demultiplexer (MDM) and data bus coupler assembly mounted on the PRCA. In addition, the Orbiter multifunction CRT display system (MCDS), the systems management computer, and a switch located on the on-orbit station standard switch panel constitute an intrinsic part of the system equipment. The MCDS includes a keyboard through which crew commands are input via the display processor to the GPC and then relayed via the bus couplers to the MDM for control of in-bay power equipment or are retransferred to the electronics assembly. PEP status data, transferred from these units to the general-purpose computer (GPC), is processed and displayed on the CRT. Figure 3.1.3-8 shows a typical PEP status display.

The PRCA is shown in Figure 3.1.3-9. It consists of six voltage regulators mounted on three Orbiter-type cold plates, three shunt limiters, a power distribution and control assembly, an MDM data bus coupler assembly, and appropriate power cables mounted to a beam support structure. As noted earlier, the PRCA remains locked in the Orbiter during the mission.

Figure 3.1.3-10 shows the PEP electrical installation in the Orbiter. Power coming down the RMS harness is routed along the PRCA beam to the voltage regulators. Power from the PRCA interfaces with the Orbiter Main A at Station 693 on the port side and with Mains B and C at Station 636 on the starboard side; all three circuit grounds are tied to Orbiter structure at these interfaces. Power cables to the main distribution assemblies from these interfaces are supplied as kit items for installation below the cargo bay liner.

The PEP electrical system is shown schematically in Figure 3.1.3-11. High voltage power from the arrays is provided via the RMS harness to the six pulse-width-modulated regulators. These regulators perform two major functions: (1) each contains a microprocessor which continuously monitors the array output and supplies control signals to assure that the peak power from the array is available to the system, and (2) they maintain a suitable output voltage (nom-

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```
┌─────────────────────────────────────────────────────────────────┐
│   XXXX/XXX          PEP CONTROL                                   │
│                                                                   │
│   PEP PWR      XXX         PWR CONN XXXXXXX    CAN ROT  1XXX  2XXX │
│                                                                   │
│   AUTO/MANUAL       XX                         MAST EXT 1XXX  2XXX │
│                                                                   │
│                GIMBAL POSITION              INFLIGHT DISCONNECT    │
│                                                  1      2          │
│   ALPHA     XXX                             ARM  X      X          │
│                                                                   │
│   BETA      XXX                             FIRE  X      X         │
│                                                                   │
│                GIMBAL RATE                                         │
│                                                                   │
│   ALPHA       XX                                                  │
│                                                                   │
│   BETA        XX                                                  │
│                                                                   │
│             REGULATOR ADJUST LEVEL                                │
│                                                                   │
│      1 XX.XX    2 XX.XX   3 XX.XX   4 XX.XX   5 XX.XX  6 XX.XX     │
│                                                                   │
│             PDB SWITCH SETTING                                    │
│                                                                   │
│      1 X 2 X 3 X 4 X 5 X 6 X                                      │
└─────────────────────────────────────────────────────────────────┘
```

Figure 3.1.3-8. PEP Control Display Format
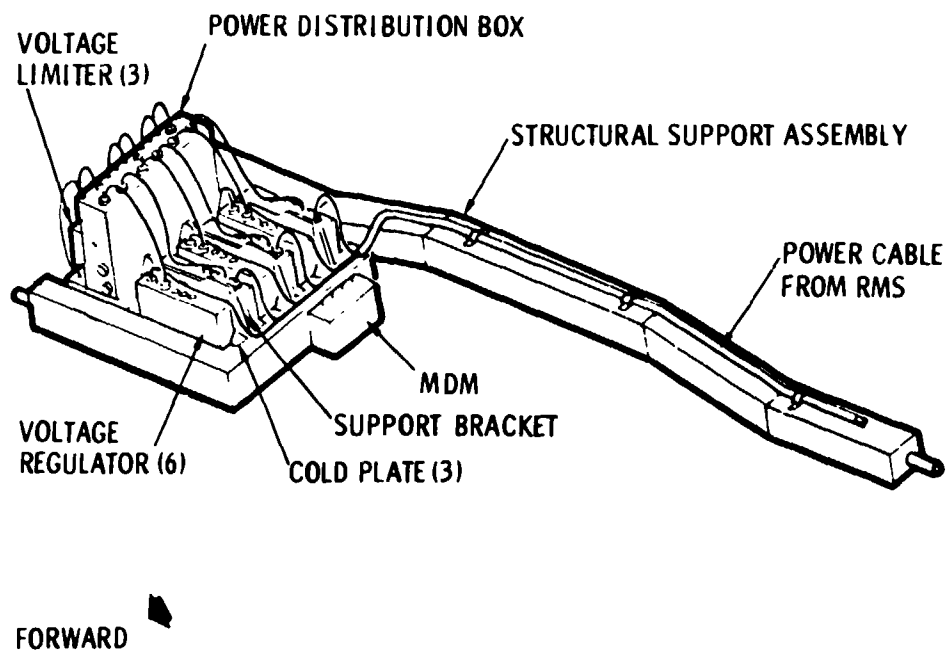
Figure 3.1.3-9. Power Regulation and Control Assembly
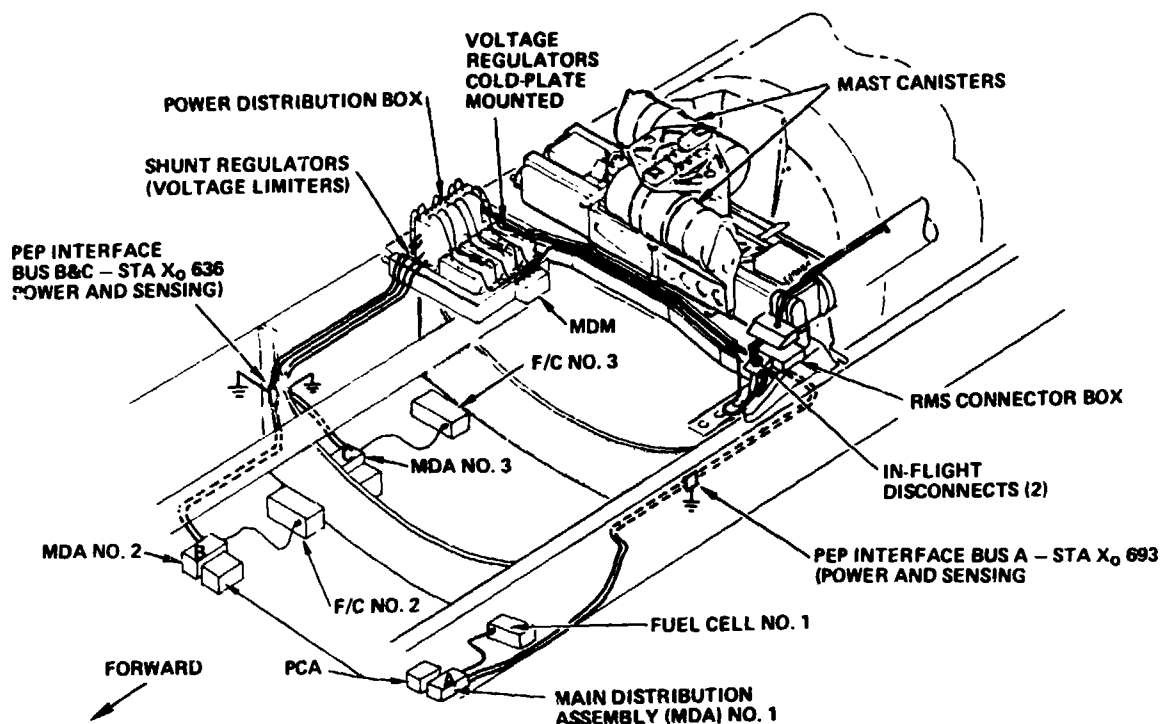
MCDONNELL DOUGLAS

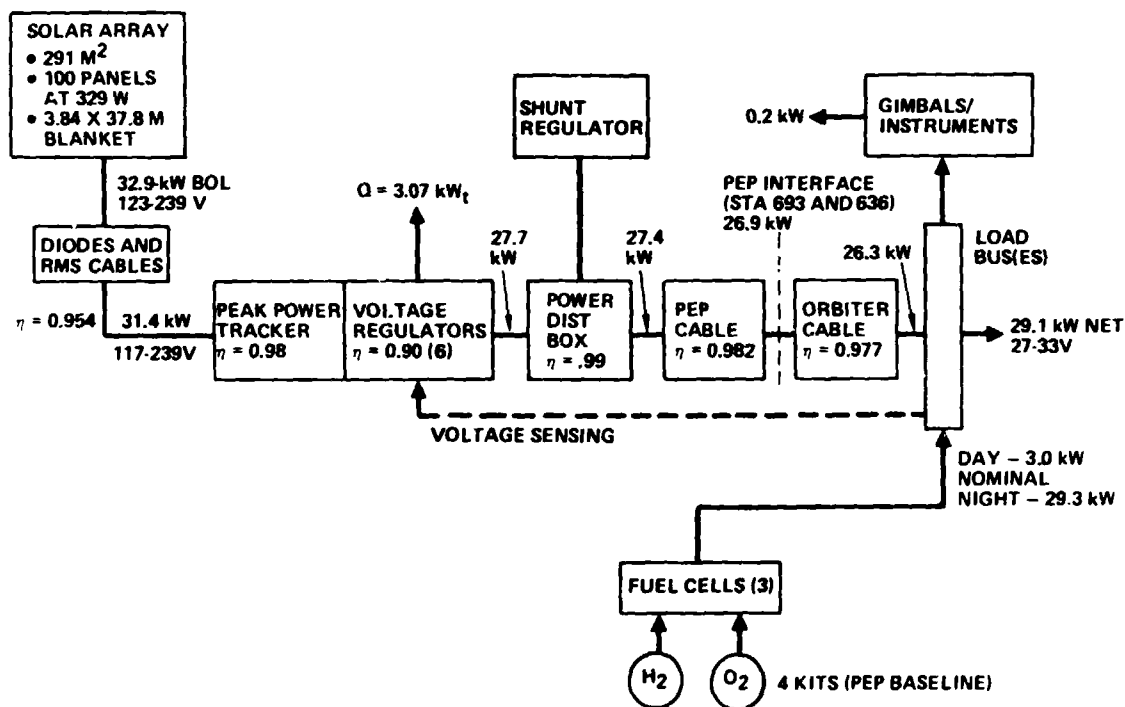**Figure 3.1.3-10. PEP Power Distribution Interfaces**

Figure 3.1.3-11. PEP Electrical Power System

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inally 32.6 V) to keep the fuel cells idling at the 1 kW needed to achieve the specified PEP duration.

Each regulator has internal overvoltage and current limiting circuitry and is provided with remote sensing capability. Output of pairs of regulators is supplied to the three Orbiter main buses.

A development PEP regulator (Figure 3.1.3-12), was constructed by MDAC using Company funds. The unit was shipped to NASA/JSC for testing on their Orbiter electric power distribution and control (EPDC) simulator and will be used to verify the method of regulating the solar array output power and the Orbiter bus interface.

The voltage regulator cold plates, which are of standard Orbiter design, are each tied into both the primary and secondary Freon 21 loops downstream of the Orbiter aft avionics bays. Quick disconnect and a jumper for use when PEP is not on board support quick PEP installation and removal.

One shunt regulator is tied to each of the three Orbiter buses via the power distribution box; in the event a failure allows a voltage regulator to supply more than 33 V to the Orbiter bus, the shunt regulator will draw sufficient current to hold the bus voltage down until the regulator can be shut down. This shutdown is accomplished automatically within 200 msec.

The deployment and mission operations of PEP are monitored and controlled at the mission specialist console in the Orbiter aft flight deck. The deployment sequence is illustrated in Figure 3.1.3-13 and timelined in Figure 3.1.3-14. The RMS grapples the array deployment assembly and moves it to the deployment assembly position where the RMS is fixed. The array is then deployed and oriented normal to the sun line. The total elapsed time is 39.1 min. if all tasks are done in series. This time can be reduced to 30.6 min. for a parallel operational alternative. Several deployment options have been analyzed which address RMS status/checkout, parallel operations, and location preferences. The retraction/storage time required is _ minutes, slightly longer, including placement of the attach trunnions back into their latch fittings.

A typical PEP operation on orbit is shown in Figure 3.1.3-15. The Orbiter is oriented with the Y-axis perpendicular to the orbit plane for an earth obser-

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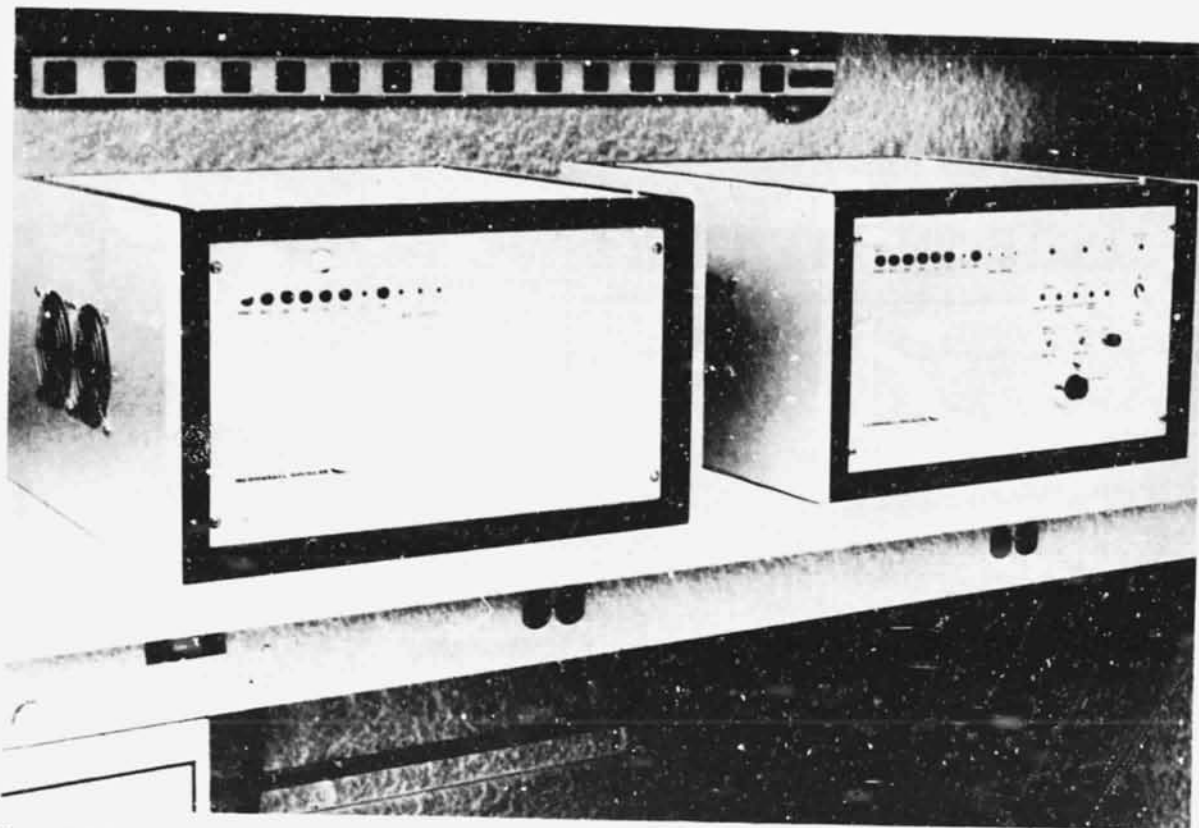
Figure 3.1.3-12. MDAC Prototype Regulator

STEP 1 – GRAPPLE ARRAY

REMOTE MANIPULATOR
SYSTEM
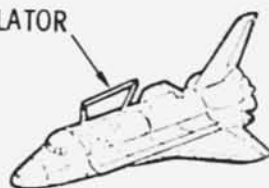
STEP 2 – UNSTOW ARRAY
DEPLOYMENT
ASSEMBLY

STEP 3 – TRANSLATE TO
OPERATIONAL
LOCATION

STEP 4 – LOCK RMS
AND DEPLOY ARRAY
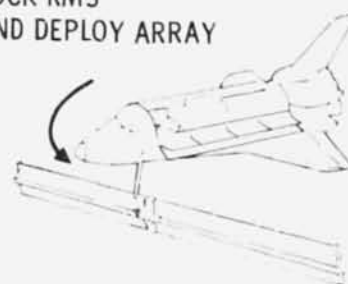
Figure 3.1.3-13. Deployment Sequence

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TIME (MIN)

OPS NO.						
	0	10	20	30	40	

1 — □ 0.5 MIN RMS C/O

2 — [3.8] RMS FROM STOW TO GRAPPLE

3 — 0.5 ☐ ENGAGE POWER CONNECTOR

4 — [1] RELEASE TRUNNION LATCHES

5 — [16.8] PEP TO FINAL POSITION

PARALLEL   SERIAL

6 — 0.5 ☐ POWER ELECTRONICS ____ 0.5 ☐

7 — [1] PEP C/O ____ [1]

8 — [2] INITIALIZE ____ [2]

9 — [4] ROTATE MAST/CANISTER [4]

10 — [1] CONFIGURE CONTACTORS [1]

11 — [2] [2] ROTATE WRIST/VERIFY

12 — [6] [6] EXTEND ARRAY

TOTAL TIME — |← 30.6 MIN →|← 8.5 MIN →|
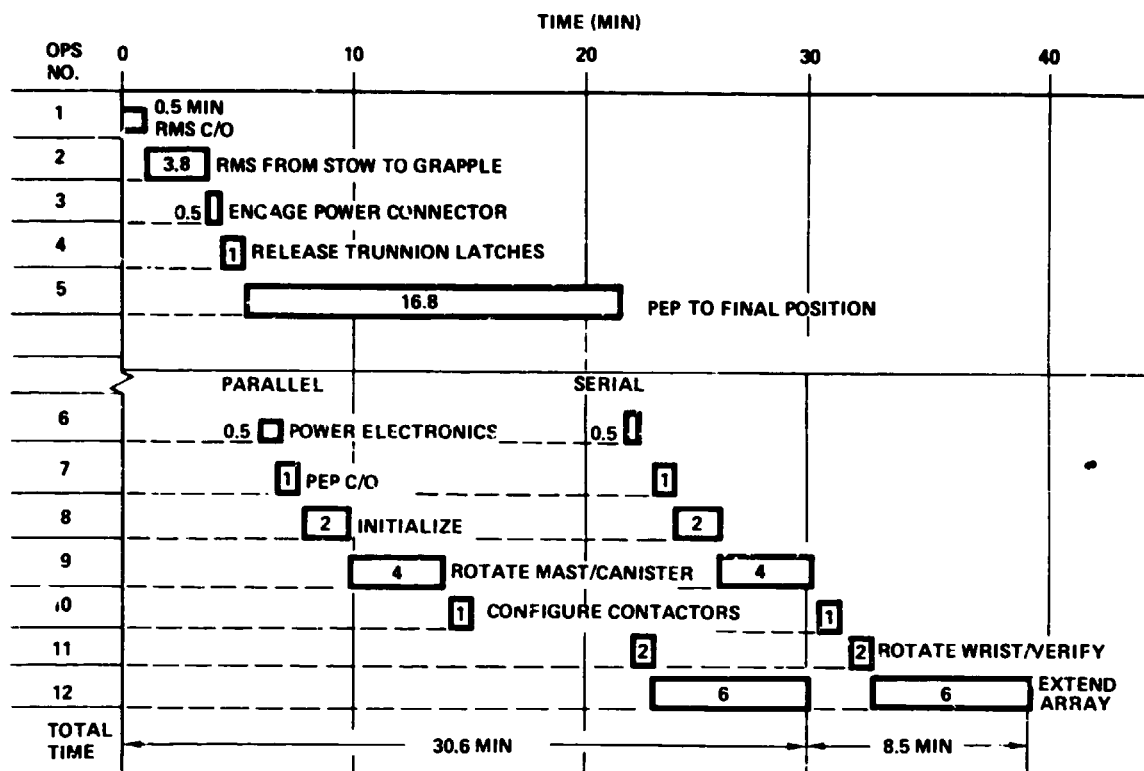
Figure 3.1.3-14. PEP Orbital Deployment Timeline (Warm RMS)
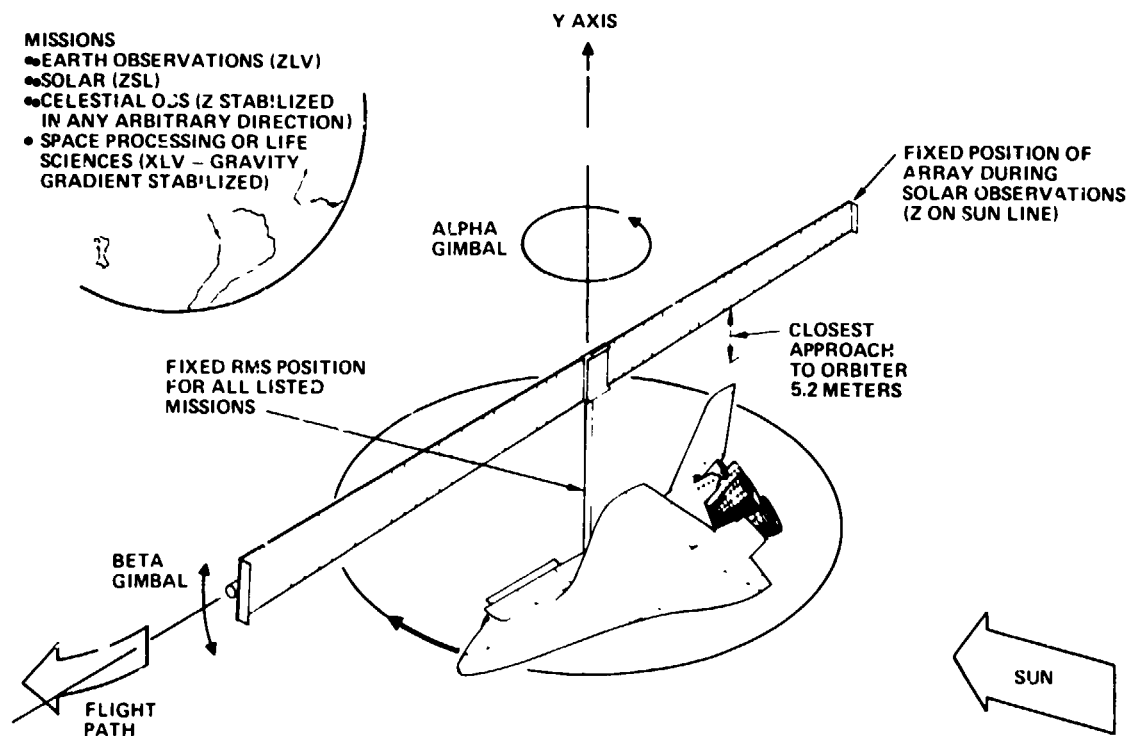
Figure 3.1.3-15. PEP Array Orientation — Y-POP Orbiter

MISSIONS
●●EARTH OBSERVATIONS (ZLV)
●●SOLAR (ZSL)
●●CELESTIAL OBS (Z STABILIZED IN ANY ARBITRARY DIRECTION)
● SPACE PROCESSING OR LIFE SCIENCES (XLV – GRAVITY GRADIENT STABILIZED)

Y AXIS

ALPHA GIMBAL

FIXED POSITION OF ARRAY DURING SOLAR OBSERVATIONS (Z ON SUN LINE)

CLOSEST APPROACH TO ORBITER 5.2 METERS

FIXED RMS POSITION FOR ALL LISTED MISSIONS

BETA GIMBAL

FLIGHT PATH

SUN

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vation mission (Z-axis aligned along the local vertical). The Alpha gimbal
axis is perpendicular to the orbit plane and thus allows relative orbital
rotation while maintaining solar alignment of the array. A zero Beta angle
condition is shown; as Beta changes with orbit regression, the operation is
the same, with the Beta gimbal slowly adjusting the array to maintain normal-
ity to the sun line.

The feasibility of the PEP design integration with the Orbiter has been
verified by layout drawings, analysis, and the definition of all the
interfaces between PEP, the Orbiter, and RMS. Figure 3.1.3-16 shows these
interfaces; see Section 4 for a more detailed description. Twenty-one distinct
interface items have been detailed in the areas of electrical, structural/
mechanical, avionics, thermal, and crew; these interface definitions resulted
from joint MDAC/RI and MDAC/Spar activities. Consistent with the design phi-
losophy of providing PEP as needed without infringing on the basic
capabilities of Orbiter, the modifications found necessary for PEP installa-
tion are minimal. They include items in the power and distribution, the
coolant loop, data bus, connector, physical attachments (electrical harness,
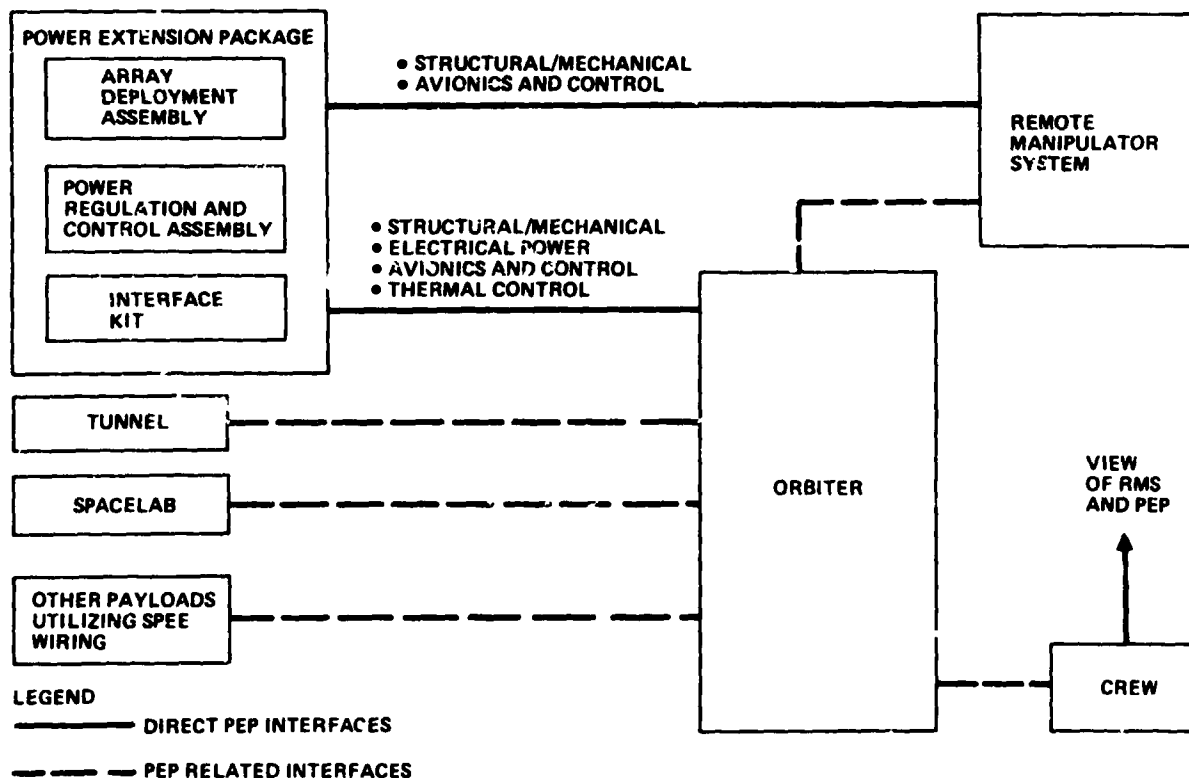plumbing, and equipment mounting), and an aft flight deck switch.

Figure 3.1.3-16. PEP Program Elements and Interfaces

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The PEP has been designed to preclude an appreciable reduction in Orbiter reliability and crew safety provisions during a PEP mission. This has been accomplished by selective redundancy, incorporation of safety provisions, and the use of manual backup by EVA for contingency operations. Table 3.1.3-2 summarizes the major safety/reliability provisions in the PEP design, and the rationale for their incorporation. More detailed treatment of PEP reliability and safety can be found in the Product Assurance Plan, Volume 8.

The need to avoid collision avoidance has been a key issue in the design and development of operational procedures for PEP. Our study has concluded that an automated collision avoidance concept is complex, expensive, and unnecessary for the PEP application. The selected approach relies heavily upon existing Orbiter/RMS equipment and procedures and incorporates the following features:

A. Strict adherence to RMS operations protocol.

B. Use of existing RMS automation to reduce probability of operator error.

C. Redundant crew monitoring by independent means of all critical RMS/PEP operations.

D. A positive means of verifying safety of the PEP array/operating position independent of (and redundant to) RMS instrumentation.

E. Use of existing RMS audible alarm to indicate any inadvertent movement of the RMS from the verified PEP operating position.

F. An array gimbal pointing system that is fundamentally fail-safe; i.e., no combination of electro/mechanical failures (other than primary structure) can cause a collision.

Details of the collision avoidance study are given in Volume 3.

Several aspects of EVA were considered in the design of the PEP. First, design criteria were considered which were necessary to enable an EVA crewman to perform PEP manual tasks such as latching/unlatching deployment/retraction, maintenance, jettisoning, and visual cueing for the RMS operator. The selected design allows for these tasks to be performed on an unscheduled and contingency basis.

The second aspect of EVA design criteria considered relates to designing the PEP to be compatible with EVA tasks which might be carried out for both PEP and other Orbiter systems or payloads. This criteria precludes damage to PEP

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Table 3.1.3-2. PEP Design Provisions for Safety and Reliability (Page 1 of 2)

Design Provision	Rationale
Redundant GPC data bus interface	Permits operation with bus, cable assembly, coupler, or MIA failure
Orbiter type or dual MDM	Provides a second control capability if MDM buffer, sequencer, or output circuit fail
Dual data bus lines from MDM to electronic assembly	Provides backup in case of MDM/electronics assembly data-transfer failure
Dual control lines from MDM to power distribution box for RMS shoulder power connector disengage signals	Permits connector separation with cable and/or relay failure
Dual control lines from MDM to actuator disengaging grapple fixture power connector	Critical control item; controls array retraction
Manual and automatic array rate/position conrol	Permits sun tracking with sun sensor/processor failure
Fusing of interface circuits	Protects Orbiter buses from possible short circuits in PEP
Shunt regulators	Provides positive backup overvoltage protection of Orbiter buses
Power contractors in power distribution box	Permits positive isolation of PEP power sources
Fuel cells on line with PEP	Complete loss of PEP will not affect mission safety
Blocking diodes in solar array series strings	Isolates a fault in any string from all paralleled strings
Dual initiator circuits for in-flight disconnects	Provides redundancy in RMS power cable separation means
Regulator sensing circuitry backup	Avoids regulator overvoltage operation if regulator remote sensing is lost
Modular isolated sources and distribution equipment	Allows for graceful degradation of PEP capability
Manual wing box top latch provision	Allows backup with EVA crewman for stowing array
Array drive motor redundancy	Backup for extension/retraction
Manual array retraction/extension	Backup for extension/retraction of arrays

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Table 3.1.3-2. PEP Design Provisions for Safety and Reliability (Page 2 of 2)

Design Provision	Rationale
Manual separation of RMS from ADA	Allows backup with EVA crewman for SPEE separation
ADA jettison provisions	Enables ADA to be jettisoned for safe crew return
Retention latch drive motor redundancy	Backup for latching/unlatching ADA in-bay
Manual retention latch/unlatch provision	Allows backup with EVA crewman
Physical separation of Freon loops	Meets Orbiter survivability criteria
PEP cold plates located under PRCA	Minimizes vulnerability of Freon loops

by EVA crews and also ensures unimpaired crew performance and safety during
EVA. Types of EVA compatibility criteria includes avoidance of exposed edges,
corners and protrusions, inclusion of handholds and translation devices, and
design to withstand crew kickoff loads. The PEP is designed based on these
criteria.

The PEP design must also consider the EVA reserved envelopes, which ensure a
corridor for the EVA crewman to access critical areas in the cargo bay. These
stay-out areas are given in "Space Shuttle System Payload Accommodation," JSC
07700, Volume IV. The reference design extends slightly into reserved EVA
envelopes but the planned installation does not prevent the crew from
performing the currently planned translations. Volume 3 of this report con-
tains the more detailed treatment of EVA considerations.

Table 3.1.3-3 presents the major system and subsystem trades and analyses con-
ducted during the study to resolve key PEP design issues. Results of this
effort are reflected in the reference PEP design. More detailed descriptions
of the trades and analyses and specific design impacts can be found in the
subsystem descriptions, Section 3.2 of this volume, and in Volume 3.

Detailed subsystem descriptions are given in Section 3.2 for electrical power,
structural/mechanical, avionics, and control, and thermal control subsystems.

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Table 3.1.3-3. Major FEP Trades and Analyses (Page 1 of 2)

Name	Purpose
Electrical	
PEP/Orbiter integration options	Evaluate interface options to idle fuel cells at 1 kW each
Regulator type and efficiency	Trade of regulator types
Regulator failure modes and effects	Identify additional design provisions required
Structural/Mechanical	
PRCA configuration	Tradeoff of 3 configurations
ADA configuration	Tradeoff of 6 configurations
Rotation and fixed mast canister	Tradeoff of the two concepts
Array wing box length analysis	Determined most suitable length of array wing box
Aluminum versus composite structure	Selected structural material for reference design
Avionics and Control	
Solar array control avionics requirements/criteria defintion	Proportional versus on-off array controller
Control system management, GPC, and array control processor interface definition	Determined number of required software modules in systems management computer and defined control and status displays
Orbiter DAP utilization evaluation	Determined loads on PEP when PRCS or VRCS are used
EMC analysis	Determined radiated field intensity impinging on PEP
Alternate solutions for PEP/RMS control and drive power wiring	Traded use of SPEE wiring versus PEP peculiar wiring harness
Thermal Control	
Alternate methods of heat rejection	Traded methods of increasing Orbiter/PEP heat rejection
Performance analysis	Determined Orbiter heat rejection capability in PEP mode
Thermal control configuration definition, active versus passive	Compare methods of cooling PEP regulators
PEP/Orbiter interface analysis	Determined suitability of Freon 21 interface
EVA considerations	Establish PEP EVA design criteria and determine EVA route intrusions

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Table 3.1.3-3. Major PEP Trades and Analyses (Page 2 of 2)

Name	Purpose
Thermal Control (Cont.)	
Avionics cooling	Determine adequacy of passive cooling for ADA avionics
General	
Reliability and safety	Verify design, identify additional design provisions required
Collision hazard elimination	Compares automated versus nonautomated methods of collision avoidance

3.1.4 Mass Properties

3.1.4.1 Reference Design Weights

The reference design weights are identified in Table 3.1.4-1 and are based on detailed subsystem and hardware definition. A description of the major element is described in the following paragraphs:

3.1.4.1.1 Array Deployment Assembly

The blanket assembly includes the panel assemblies and the harness provisions with weight supplied by subcontractors. The panel assembly weights were developed for a 32.9-KWe array at BOL with 100 active panels. The total panel area is 3,127.4 ft^2 (290.7 m^2). Harnesses include the feeder harness, which is aluminum.

The wing box assembly is made up of the wing boxes, blanket tension mechanism, and the mast interface linkage assembly. The wing boxes and interface linkages weights as supplied by the subcontractor were a composite design and these values were reworked using aluminum. The four blanket tension mechanism weights were also supplied by the subcontractor.

The solar array mast assembly weights were developed by the subcontractors for a 125.3-ft (38.2 meters) mast with 200 ft-lb ultimate bending moment. The assembly includes the canister and mast drive actuators.

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Table 3.1.4-1. Reference Design Weights

| Description | Location | | Kits | | |
| | ADA | PRCA | RMS harness | Power distribution harness | Thermal control |
				Payload bay*	
Solar array	1,031	–			
Blanket assembly	615				
Solar array wing box assembly	160				
Solar array mast assembly	256				
Structure/mechanical	232	68			
Solar array support structure assembly	126	–			
Power regulation equipment support structure assembly	–	68			
Solar array canister support mechanism	33	–			
Z-axis solar array drive gimbal assembly	73	–			
Power distribution and regulation	54	534	–	110	–
Power-distribution equipment	22	75		–	
Voltage-regulation equipment	–	432		–	
Distribution cables	32	27		110	
RMS power cable assembly	–	–	101	–	–
Thermal control	–	34	–	–	10
Avionics	57	32	–	3	–
Subtotal (lb)	1,374	668	101	113	10
*Total weight (lb)			2,266		

*Orbiter scar; power distribution (37 lb) and thermal (9 lb)

The structure/mechanical subsystem contains the support structure assembly which is open-truss beam design with a three-trunnion support system. The canister support (pivot assembly) includes the canister suspension, lockout mechanism, and stowed position latches. The final assembly is the gimbal assembly made up of the grapple fixture the umbilicals, and the Alpha and Beta gimbal drive assemblies.

Power distribution and regulation subsystem on the ADA consists of two diode boxes plus the power distribution wiring harnesses. The diode box weights are based on a component listing, and the necessary weights for wiring and housing.

The avionics susbsystem consists of a sun sensor, signal processor, array control electronics with associated instrumentation, and wiring.

3.1.4.1.2 Power Regulator and Control Assembly

The PRCA structure support assembly is an aluminum beam design with a three-trunnion interface with the Orbiter.

The power distribution and regulation subsystem includes the power distribution box, three shunt voltage limiters, and six MDAC-designed voltage regulators. The weights for the above were all derived from detail component lists plus allowances for wiring and housing of each assembly. The distribution cables includes three harness assemblies; they are the separation device to the voltage regulator, the voltage regulators to the power distribution box, and the power distribution box to the shunt voltage limiter.

The thermal control subsystem weights are based on a cold-plate design to match the required equipment mounting area. The total cold-plate area is 8.9 ft^2 (0.83 m^2). The remaining portion of thermal control includes the lines, disconnects, and fluids necessary to complete the installation.

The avionics subsystem on the PRCA consists of the multiplexer/demultiplexer plus wiring to the RMS interface.

3.1.4.1.3 Kits

There are four items in this category. The first is the RMS harness kit which runs the length of the RMS from the PRCA to the separation device near the RMS shoulder. The weights were extracted from SPAR inputs. The second is the power

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distribution harness from the power distribution boxes on the PRCA to the interface panels. Two on the standard side (-Yo) at station (Xo) 636 and one on the port side (+Yo) at station (Xo) 693. The third is the thermal control with the fourth being the wiring from the MDA to the data bus coupler.

3.1.4.1.4 Scar

The scar weights are, as were all cable weights, based on estimated lengths and cable sizes, with allowances for connectors and supports. R.I. supplied the cable lengths for Orbiter scar weights. The total harness weight for scar is 37 lb (16.8 kg). The thermal scar weight of 9 lb (4.1 kg) was supplied by R.I.

3.1.4.2 Center of Gravity and Inertia

The CG's and inertia values are referenced to Figure 3.1.4-1 and use the Orbiter coordinate axes and stations. The data is summarized in Table 3.1.4-2 using the same reference for all configurations with the RMS CG interface being $X_o = 715$, $Y_o = -38$, and $Z_o = 471$.

Tables 3.1.4-3 and 3.1.4-4 illustrate two typical PEP missions with impact for weight and the Xo center of gravity. The Spacelab II mission (Table 3.1.4-3)
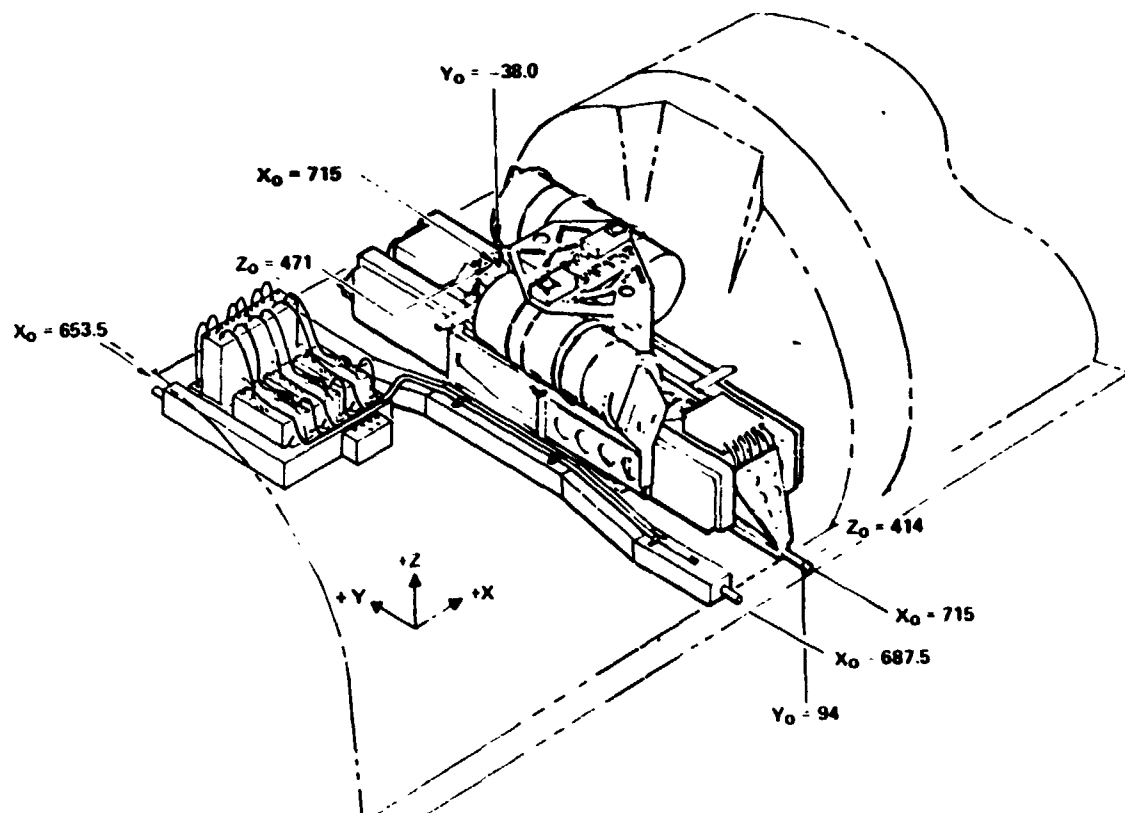

Figure 3.1.4-1. PEP Coordinate Axes and Stations

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Table 3.1.4-2. PEP Center of Gravity and Inertia

Data Configuration	Weight (lb)	Center of gravity (inches)			Moment of inertia (slug-ft2)			Product of inertia (slug-ft2)		
		X_o	Y_o	Z_o	I_x	I_y	I_z	I_{yx}	I_{zx}	I_{xy}
ADA – array stowed	1,374	716.7	-2.1	448.3	159	102	170	-6.7	-5.0	-27.8
ADA – array deployed	1,374	723.8	-2.1	445.4	380	145,134	145,400	-8.5	-26.7	3.2
PRCA	668	667.7	-52.6	409.8	151	14	151	-4.1	4.1	18.8
Kits and Scar	224*	784.6	31.7	386.7	362	1,527	1,611	132.7	438.8	412.8
Retention provisions	85	701.7	60.8	408.1	94	22	117	-1.1	1.1	-8.1
PEP – launch config.	2,266	708.9	-13.6	430.9	1,278	2,434	2,830	203.5	442.7	828.2
PEP – total	2,351	708.7	-11.0	430.1	1,480	2,467	3,046	172.4	446.7	810.6

*Total Scar weight 46 lb

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Table 3.1.4-3. Spacelab II Weights and X_o Center of Gravity

Event	Landing		Launch/abort	
Element	Weight (lb)	X_o	Weight (lb)	X_o
Spacelab II	26,203	967.8	28,492	973.3
PEP	2,226	708.9	2,266	708.9
Retention provisions	85	701.7	85	701.7
Subtotal	28,553	947.2	30,842	953.2
Ballast	500	1,301.0	500	1,301.0
Total	29,053	953.3	31,342	958.7
X_o Margin (in.)	+ 4.0		0.0	

Table 3.1.4-4. Spacelab III Weights and X_o Center of Gravity

Event	Landing		Launch/abort	
Element	Weight (lb)	X_o	Weight (lb)	X_o
Spacelab III	28,683	1,001.2	30,599	1,005.1
PEP	2,266	708.9	2,266	708.9
Retention provisions	85	701.7	85	701.7
Subtotal	32,033	979.1	32,949	984.0
Ballast	-	-	-	-
Total	31,033	979.0	32,949	984.0
X_o Margin (in.)	+21.6		+19.6	

is less than the 32,000-lb (14,500 kg) landing/abort limit with approximately 500 lb (227 kg) of ballast being required at Station 1301 to bring the Xo CG for shuttle cargo. Spacelab III (Table 3.1.4-4) Xo CGs are well within the allowable cargo CG limits. The launch/abort weight of 32,949 lb (14,977 kg) is not a problem because the planned landing weight is 31,033 lb (14,106 kg) with PEP. A possible source of ballast could include the loading of extra Orbiter OMS propellant. For Spacelab II type missions, such ballast would eliminate the addition of special ballast-mounting provisions.

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3.1.4.3 Weight-Saving Options

Table 3.1.4-5 is a list of potential weight-saving options. Each item should be also evaluated first for cost and then in order of its respective dollars/pound to determine the most cost-effective options.

Table 3.1.4-5 Potential Weight-Saving Candidates

Composite structure	104	
Wing box		27
Mast/canister		23
Canister pivot assembly		9
ADA		23
PRCA		16
Mast interface linkage		6
Solar array	179	
Cell thickness (6 to 4 mils)		59
High-efficiency OCLI cells		120
Miscellaneous	10	
Eliminate RMS cable shroud		10

3.1.4.4 Summary

The December 1978 weight of 2,010 lb (914 kg) and the primary reason for change are listed in Table 3.1.4-6. The primary elements of change were the strongback design, subcontractor definition of the wing boxes, incorporation of the diode boxes, and redefinition of the voltage regulators.

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Table 3.1.4-6. Initial to Reference Design PEP Weight Comparison

	Initial weight (lb)	Reference design weight (lb)	Δ Weight (lb)	Remarks
Solar array	955	1,031		
Blanket assembly	580	615	+35	Initial from Seps – subcontractor redefined
Solar array wing box assembly	110	160	+50	
Solar array mast assembly	265	256	-9	Miscellaneous
Structure/mechanical	260	300		
Solar array support struct assembly	93	126	+33	Integral wing box to strongback design
Power reg equipment support struct assembly	61	68	+7	Added design definition
Solar array canister support mechanical	33	33	-	-
Z-axis solar array drive gimbal assembly	73	73	-	-
Power distribution and regulation	552			
Power dist equipment	87	97	+10	Incorporation of diode boxes and elimination of J-boxes
Voltage reg equipment	339	432	+93	Added design definition of voltage regulators
Distribution cables	126	169	+40	New power dist equipment definition and incorporation of R.I. data
RMS power cable assembly	90	101	+11	Spar definition
Thermal control	61	44	-16	Smaller cold plates
Avionics	92	92	-	-
Total weight (lb)	2,010	2,266	+256	

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3.2 SUBSYSTEM DESCRIPTION

3.2.1 Electrical Power Subsystem

The function of the PEP electrical power subsystem (EPS) is to generate
electrical power on-orbit from incident solar energy and deliver 26.82 kW
gross to the three Orbiter electrical bus systems at voltages compatible with
the Orbiter electrical requirements and the fuel cells when they are operating
in the range of approximately 1 to 12 kW each. The basic elements of the
system are a gimballed solar array, voltage regulators, and miscellaneous
power cables and distribution equipment.

3.2.1.1 Electrical Power Subsystem Requirements

The principal EPS requirements are summarized in Table 3.2.1-1. The differ-
ence between gross and net output power is 0.200 kW of parasitic power allowed
for operation of PEP. PEP power has frequently been quoted at the load buses
or the payload interfaces. The difference between this location and the PEP
interface with the Orbiter is Orbiter cable and distribution losses, which
correspond to an efficiency of 97.7%; PEP output should be specified at the
PEP interface where the gross requirement is 26.82 kW (the system actually
delivers 26.90 kW as will be discussed later).

The PEP power output must be delivered in three isolated blocks, each of
which operates in parallel with an Orbiter fuel cell. The loads associated
with each of these three blocks are not well-known and will change from
mission to mission. Consequently, it is necessary to have the capability to
reallocate between missions varying proportions of the total PEP output to
each of the three blocks in increments of 1% of the total (e.g., 0.2682 kW).
Each of the three blocks must be designed to handle more than 38.1% of the
total (e.g., 0.381 x 26.82 kW = 10.22 kW).

The steady-state voltage regulation should not exceed 33.0 V at the load
interface in the event of failure of the normal voltage regulators. Nominal
voltage regulation is 32.5 to 32.7 V when PEP is sharing with the fuel cells
operating at 1.0 kW each. Voltage regulation for fuel-cell sharing is in
the range of 27 to 32.5 V when PEP output capability is exceeded. The PEP
life requirement is 80 missions over a 10-year period at a typical duration
of 14 days each; hence, the on-orbit life requirement is 3.07 years.

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Table 3.2.1-1. PEP EPS Requirements (Page 1 of 2)

Item	Requirement
Orbit altitude, km	185-1,110; 407 nominal
Orbit inclination, deg	28.5-104; nominal 28.5
Sun angle, β, deg	0 to ±127; 50 nominal
Array shadowing, %	0-100 at orbit rate; 0 nominal
Output power, gross/net kw	
● Load buses	26.20
● PEP interface	26.82
● Modularity/flexibility	
– Number of isolated blocks	3 minimum
– Capability of each block	38.1% of total nominal
– Block adjustment increment	1% of total
Maximum parasitic power, kW	0.200
Steady-state voltage regulation[1], V	
● Maximum (failure modes)	33.0
● Voltage regulator mode	32.5-32.7
● Power tracking and current limiting modes	27-32.5
● Regulation adjustment increment (remote)	0.02 V
Transient voltage regulation, V	+0.3, -0.5 recovering to within steady-state regulation band in 5 msec
Life	
● Operational period, years	10
● Missions per year	8
● Average mission duration, days	14
Storage	
● VAB, hours	960
● Hangar S or OPF, years	7
Array deployment/retraction/lockup	
● Time, minutes	6 maximum
● Power (2 wings)	8.0 A at 18-33 V
Orbiter compatibility	Minimum impact and scar

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Table 3.2.1-1. PEP EPS Requirements (Page 2 of 2)

Item	Requirement
RMS power harness	Must not restrict RMS functions Removable/installable in field In-flight separation provisions at end effector and base of shoulder
Power distribution	Compatible with Orbiter three bus/ fuel cell system
Power circuit grounding	Single-point ground in Orbiter bay
EMI	Meet Orbiter specs
Instrumentation	Provide sufficient information to the Orbiter AFD displays to allow manned supervision of the EPS

3.2.1.2 Electrical Power Subsystem Performance

The PEP EPS requirement is to deliver 26.82 kW minimum to the PEP/Orbiter interface. The capability of the next integral number of array panels per wing yields a PEP system power capability of 26.90 kW to the interface. This is sufficient to support a 14-kW Orbiter load and a 15-kW payload requirement (net) when the PEP output is combined with 1.0 kW from each of the three fuel cells.

The PEP system performance is depicted in Figure 3.2.1-1 for several values of β. The design point case of 26.90 kW is for β = 50 deg at the midpoint of the illuminated period of the orbit. The power at β = 90 deg is continuous at 27.97 kW. The minimum at β = 0 deg is 25.55 kW; the design point value is approximately the average value for the illuminated period of this orbit. The 26.90-kW design point value degrades to approximately 25.0 kW EOL for the ETR version PEP at the end of the 10-year operational period (3.07 years on-orbit).

The PEP voltage regulation performance conforms to the requirements discussed above. Means are provided for protecting the Orbiter buses from overvoltage or short circuits which may develop in the PEP EPS.
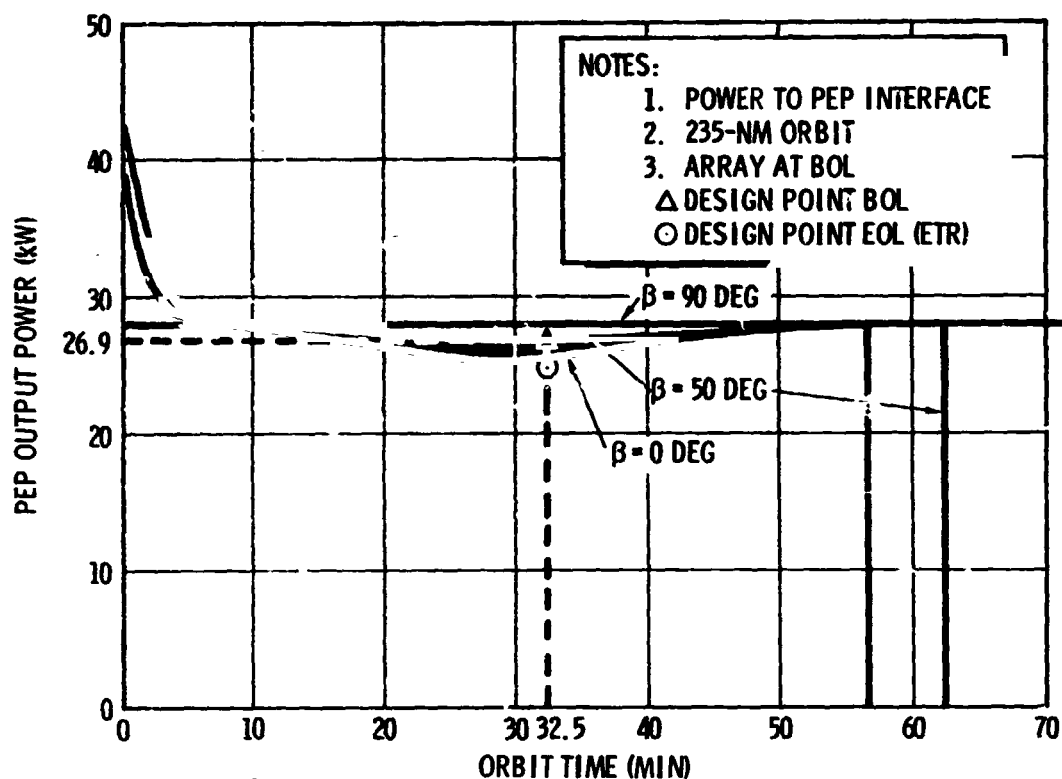
Figure 2.2.1-1. PEP Power Output Performance

Power available to the load buses can be reallocated up to 10% above or below nominal on a mission-by-mission basis to accommodate changes in bus loads.

3.2.1.3 Major Trades and Analyses

3.2.1.3.1 PEP/Orbiter Integration Options

A tradeoff analysis was conducted to evaluate alternative options of electrically interfacing with the Orbiter to control the fuel cells to 1.0 kW each while staying within the orbiter maximum voltage limitation. The performance of the Orbiter fuel cells and the PEP system (voltage regulator and shunt limiter) is shown in Figure 3.2.1-2. The nominal operating point is the intersection (at a common voltage) of the fuel cell curve and the dashed line representing the midpoint of the voltage regulator operating band, fuel cell idle power is adjusted by raising or lowering the regulator band. The midpoint of the voltage regulator band must be 0.4 V below the maximum allowable steady-state voltage (33.0 V in the example) to provide a distinct operating region for the shunt limiter.
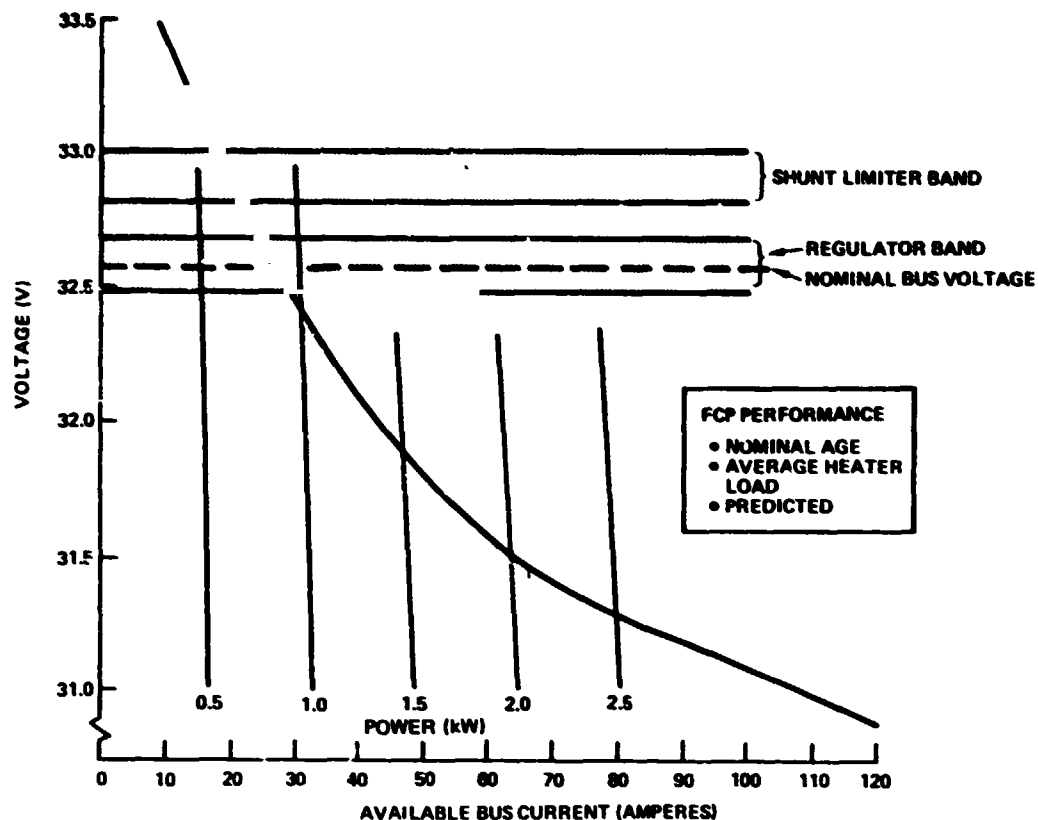
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Figure 3.2.1-2. PEP and Orbiter Fuel Cell Voltage Regulation for 33 V Bus Limit

A summary of the trade results is presented in Table 3.2.1-2 for the three
key options; the voltages in parentheses are the maximum allowable Orbiter
load voltages. The J-box/splice, aft PCA 4 scheme, is a variant of the scheme
derived in the previous phase of the PEP study; an advantage of this approach
is that modifications to the main distribution assembly (MDA) are not required.
However, the mission duration disadvantages were also recognized late in the
previous study phase and utilization of the MDA's was discussed with JSC
personnel at the November 1978 final review. Rockwell personnel were queried
in early January about the possibility of tying the power and sensor leads
into the Orbiter at or near the fuel cells, this scheme was subsequently
adopted, by modifying the MDA's, and is the current Interface Definition
Document (IDD) approach. The maximum allowable Orbiter load interface
(bus) voltage was selected to be 33.0 V because of the mission duration
advantages and the system complexity disadvantages of the alternative diode
approach (see Volume 3, Section 2.6, for more information on the diode and
other options).

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Table 3.2.1-2. PEP/Orbiter Integration Option Summary

Item	J-box/splice aft PCA 4 (32 V)	IDD (32 V)	IDD (∿32.7 V)
FCP power (average), kW	2.04	1.66	1.0
PEP power output (PDB), kW	24.2	26.0	∿28.1
Mission duration	15.6	17.0	19.8
Weight, lb	714	776	776
Array blanket cost, $ M	6.08	6.61	7.14

Note: Bus A, B, and C at 4.67 kW each plus 15 kW to Spacelab

As the FCP average power goes down, the PEP system output power, and hence solar array cost, rises as shown in Table 3.2.1-2, for a constant 29.0-kW load requirement. The array blanket cost represents the blanket recurring cost for a single PEP system. The weight includes the blanket weight and a delta (Δ) for wire and related distribution equipment.

In summary, this trade resulted in the selection of a 33.0-V bus voltage limit and support for the selection of the IDD approach, in conjunction with Rockwell's analysis of interface alternatives.

3.2.1.3.2 Regulator Type and Efficiency

A trade study of different regulator types was performed and the pulse-width-modulated buck type was selected for the PEP application. This type of regulator with recently developed proprietary MDAC low loss snubbers approaches 92% overall efficiency at the design operating point. The reference EPS was sized using an efficiency of 90% as shown on the design requirements summary in Table 3.2.1-3. The higher indicated efficiency using the new snubbers will significantly reduce regulator heat rejection to the Orbiter coolant loop.

Regulator design in reasonably straightforward compared to other types. A pair of 2.5-kW buck regulators was developed under Company funds for use by JSC in simulated PEP system tests on their electrical power distribution

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Table 3.2.1-3. PEP Power Regulator Design Requirements

Rated input voltage, V_{IN}	117 volts
Maximum input voltage, V_M	239 volts
Rated output voltage, V_O	33 volts
Voltage regulation (at Orbiter main bus)	32.5 to 32.7 volts
Maximum output voltage ripple	0.1 volts peak to peak
Rated output current, I_O	145 amperes
Rated output power, $V_O \times I_O$	4.8 kilowatts
Efficiency at rated V_{IN}, V_O and I_O	90%
Maximum output current, I_M	160 amperes
Maximum output power, $V_O \times I_M$	5.3 kilowatts
Loss of remote voltage sensing circuit	Revert to internal reference and operate at lower output voltage
Peak power tracking	Track solar array peak power point within 2% for output currents less than regulator current limit set point
Electromagnetic interference (EMI)	Meet Orbiter specifications
Heat rejection	To cold plate at TBD $^{\circ}$C

and control (EPDC) breadboard. The low loss snubbers became available later
and will be considered for use in flight units to increase overall system
efficiency. Microprocessor-based control is used to perform peak power track-
ing and other computational/programmable control and protection functions.
Extremely good regulation is achieved. This is particularly important because
the PEP design requires close control of voltage at the Orbiter fuel cells
during sunlight operation to maximize mission durations.

3.2.1.3.3 Regulator Failure Modes and Effects (FMEA)

An analysis was performed to determine the effects of regulator failure modes
on EPS performance. Emphasis was placed on analyzing failures which could
lead to bus overvoltage and on incorporating the necessary provisions in
regulator/EPS protection design to prevent bus voltage from exceeding
specification limits.

The regulators, together with their dedicated solar array sources and paralleled fuel cell on the output, were modeled and analyzed for transient behavior using the SPICE computer program. Almost all failure modes identified resulted in fail-operational conditions. For cases involving the loss of one of the two regulators on a bus, the paralleled fuel cell must make up for the loss of power, with corresponding reduction in mission duration.

Section 2.7 of Volume 3 provides further data on MDAC regulator work.

3.2.1.4 Electrical Power Subsystem Description

The function of the PEP electrical power subsystem is to generate electrical power on-orbit from incident solar energy and deliver 26.82 kW gross to the three Orbiter electrical bus systems at voltages compatible with the Orbiter electrical requirements and the fuel cells when they are operating in the range of approximately 1 to 12 kW each. The basic elements of the system are a gimballed solar array, voltage regulators, and miscellaneous power cables and distribution equipment.

3.2.1.4.1 Subsystem Description/Power Flow

Figure 3.2.1-3 shows the major elements of the system and the power flow through the system, which delivers 29.08 kW of net power to the Orbiter buses and payloads. The PEP gross power is 0.200 + 29.08 - 3.0 kW = 26.28 kW to the load interfaces or 26.90 kW at the PEP interface connectors, where 0.200 kW is an allowance for the PEP parasitic load and 3.0 kW is the total contribution of the three Orbiter fuel cells. The nominal power and voltage at various locations in the system are shown on the diagram. The array output (input to the array isolation diodes, diode assembly box) is 32.88 kW or 16.44 kW for each wing. System voltage is controlled by the voltage regulators to the required 32.5 to 32.7 V at the load buses/load interface by means of remote sensing cables; this system compensates for voltage losses between the regulators and the load interfaces.
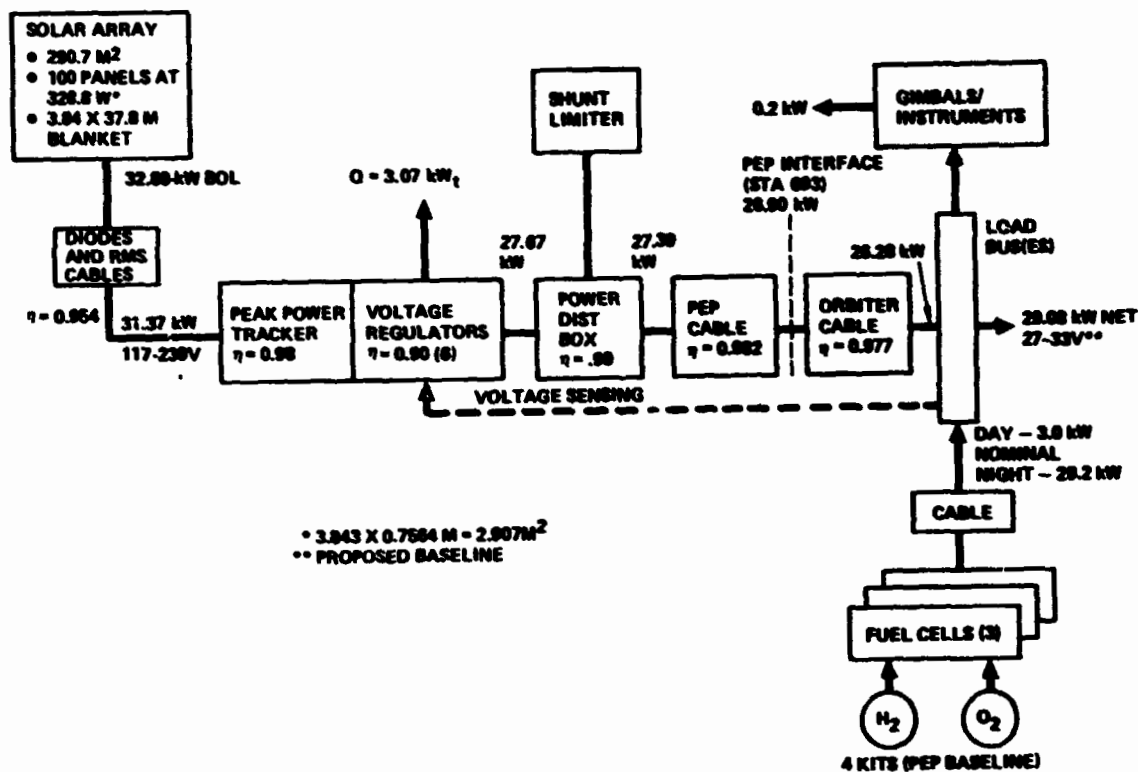
Figure 3.2.1-3. PEP Electrical Power System

The function of the peak power tracker feature of the voltage regulator
is to control the operation of the array at its peak power point when load
demands exceed array capability (e.g., during overloads or array shadowing).
When this occurs, the regulator output voltage falls below the regulation
band, permitting the fuel cell output power to rise to meet the load
demand. Peak power tracker efficiency losses relate to its inability to
perfectly track the peak power point; these losses result in the need
to oversize the array but do not result in regulator heat rejection.

3.2.1.4.2 Distribution Interfaces/Schematics

Distribution interfaces between PEP and Orbiter are indicated on Figure
3.2.1-4. Power and voltage sensing interfaces with Main A/FC 1 are shown
at Station X_o 693 on the port side. Interfaces with Main B/FC 2 and Main
C/FC 3 are at Station X_o 636 on the starboard side. Power is supplied to
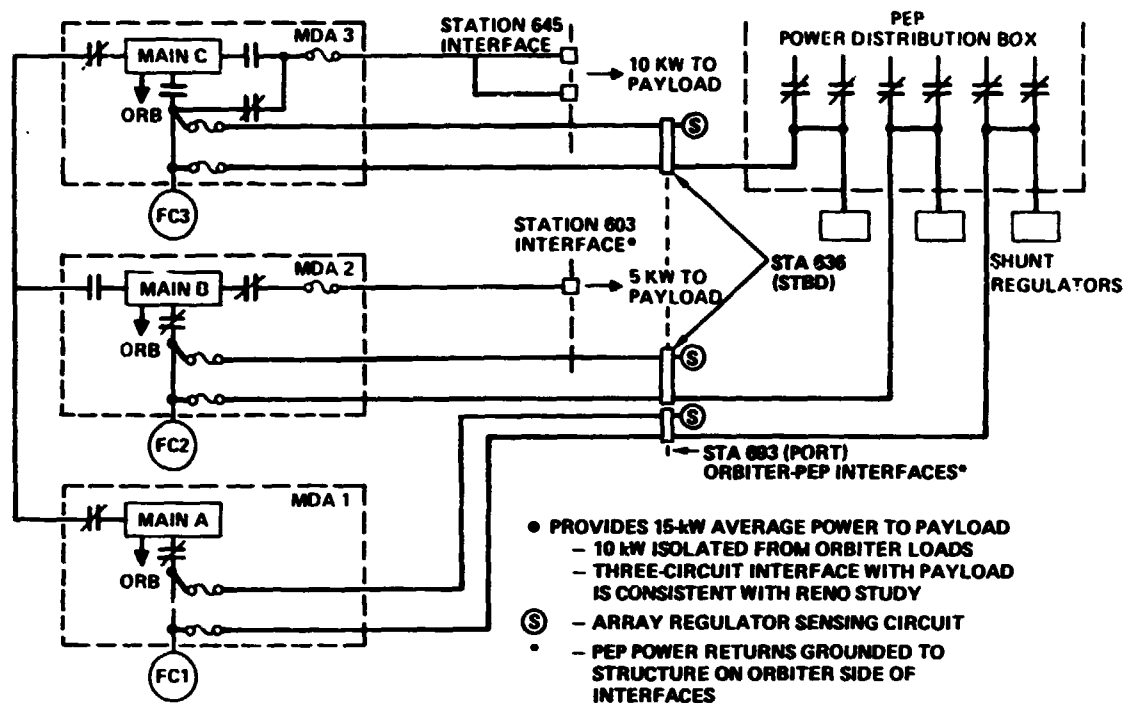each interface from an isolated bus in the PEP power-distribution box.

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Figure 3.2.1-4. Power Distribution Interfaces

PEP power into MDA 1 (lower left portion of figure) is paralleled with fuel cell No. 1 to supply Orbiter Main A loads directly and Main C loads via the tie bus to MDA 3. Power into MDA 2 supplies Orbiter Main B loads in parallel with up to 5 kW to payload via a new circuit from MDA 2 to payload interface at Station X_o 603. PEP power into MDA 3 is paralleled with fuel cell No. 3 (isolated from Main C) and fed directly to the payload over existing circuits, with the interface at Station X_o 645. These circuits provide the payload with up to 10 kW average power from dedicated sources in both the Orbiter and PEP.

Figure 3.2.1-5 shows routing of power and voltage sensing cables from the PEP power-distribution box to the Orbiter interface connector panels at Stations X_o 693 and X_o 636. The power return cables are shown grounded to structure on the Orbiter side of each interface. The positive polarity cables and the twisted-shielded pair voltage-sensing cables continue on to their designated MDA's.
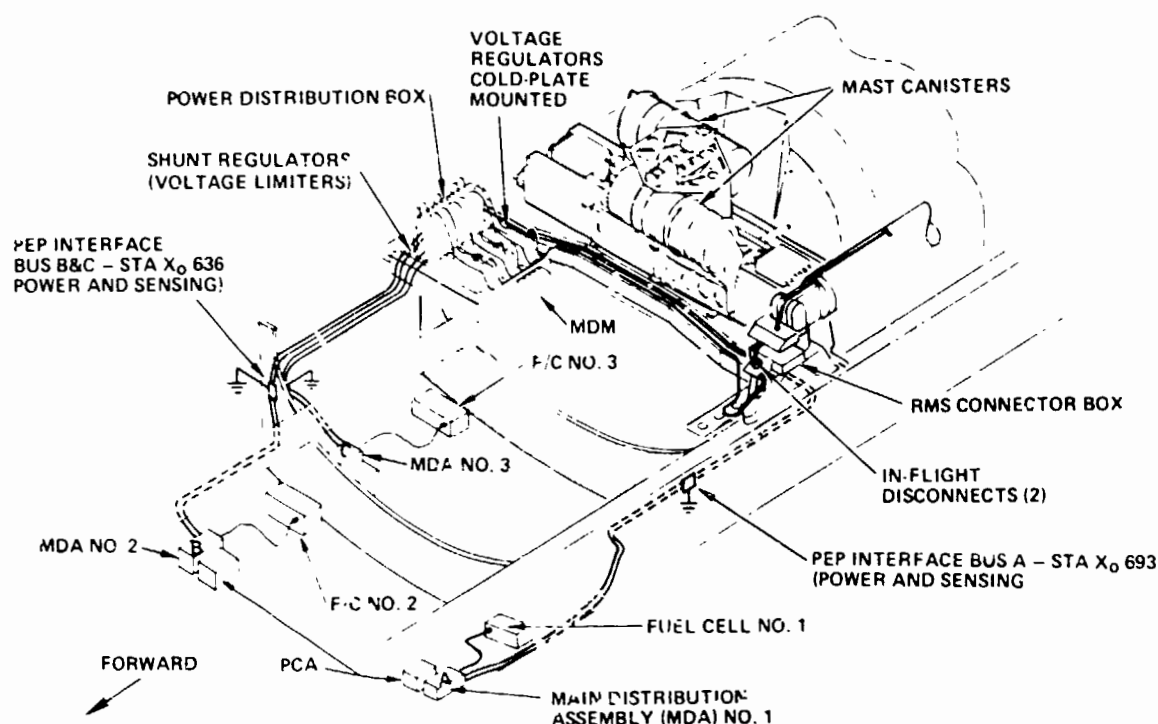
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Figure 3.2.1-5. PEP Power Distribution Interfaces

3.2.1.4.3 ADA/RMS/PRCA/Orbiter Harness Diagram

Figure 3.2.1-6 gives an overview of the PEP EPS in elementary diagram form. All major hardware items and cables/harnesses are shown. Reference to avionics and control in the gimbal assembly block is for mate-demate control of the umbilical actuator. Similarly, the reference in the PRCA block is for circuits to the redundant separation initiators (nonpyrotechnic) in the two RMS-PRCA interface connectors (in-flight disconnects).

A summary of power-distribution cable run lengths, number and size of wires in each run, and estimated weights from the array base in the wing boxes to Orbiter and payload interfaces is given in Table 3.2.1-4.

3.2.1.4.4 Solar Array Description

A summary of the characteristics of a representative solar array wing concept (MDAC reference design based on an LMSC example) is presented in Figure 3.2.1-7. The PEP array consists of a pair of such wings, each of which is folded accordian style into a box for launch and deployed on orbit by the extension/retraction mast. The power (16.4 kW, BOL, and approximately 15.2 kW, EOL,
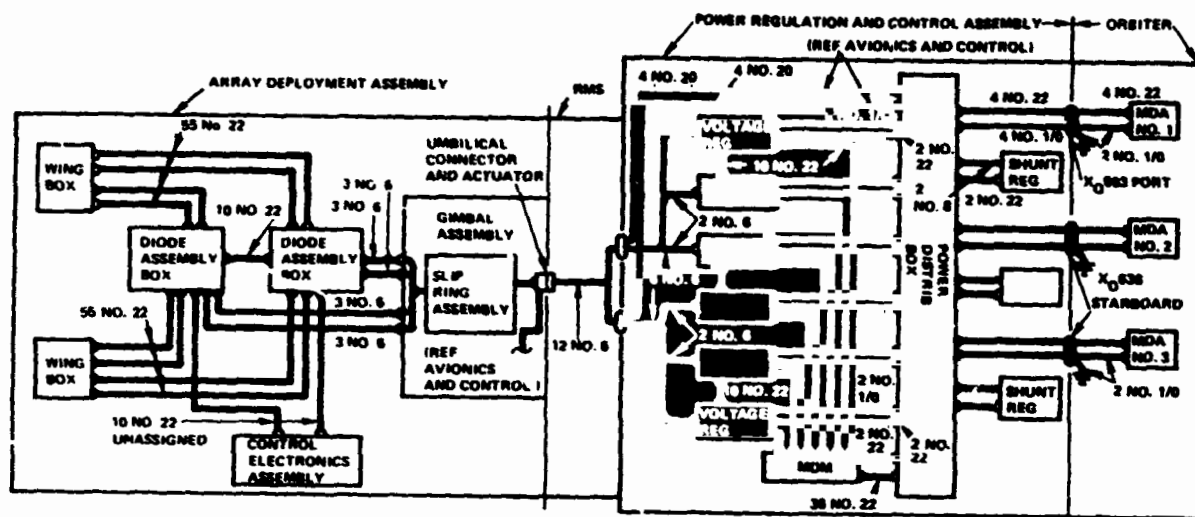
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Figure 3.2.1-6. PEP EPS Elementary Diagram

for the PEP used at ETR) is generated by 50 solar cell panels and conducted by a wire harness down each edge of the blanket. Each of the 50 panels is folded in the middle and joined to the next panel by a piano hinge. Blanket tension is uniform across the blanket; it is provided by two tension bottom springs. Two guide wires run the length of the panel to assure proper panel folding and location in the array storage container during retraction.

The solar cell assembly and blanket assembly is shown in Figure 3.2.1-8. The solar cells are of the high efficiency (12.8% nominal, 28°C) hybrid type, 2 x 4 cm in size with a back surface reflector and a base resistivity of $1-3\Omega$ -cm. The cell voltage at maximum power is 0.45 V and the power output is 17.4 mW/cm^2 at 28°C. Welding is employed in the example described above to connect the cell assembly to the substrate; repair is done by soldering.

Table 3.2.1-4. Power-Distribution Cable Weights

Cable run	Run length (ft)	Number of wires	Wire gage	Weight (lb)
Array base to diode assembly boxes	2	440 equiv (flat cables)	Varies	9
Between diode assembly boxes	11	10	22	1
Diode assembly boxes to gimbal assembly (excludes slip ring assembly)	7 4	6 6	6 6	6 4
Along the RMS	70	12	6	87
RMS to voltage regulators	15	12	6	18
Voltage regs to power dist box (PDB)	2	12	1/0	10
PDB to shunt regulators	3	6	8	2
PDB to Sta 693 interface	20	4	1/0	32
Sta 693 to structure ground	2	2	1/0	2
Sta 693 to Main A/FC1	7	2	1/0	5
PDB to Sta 636 interface	6	8	1/0	19
Sta 636 to structure ground	2	4	1/0	3
Sta 636 to Main B/FC2	20	2	1/0	16
Main B to Sta 603 payload interface	5	4	1/0	8
Sta 636 to Main C/FC3	6	2	1/0	5
Main C/FC3 to Sta 645 payload interface		(existing)		
Total cable weight				227

3.2.1.4.4.1 <u>Solar Array Options and Performance</u>. LMSC and TRW are designing PEP solar array systems as depicted in the artist's concept of Figure 3.2.1-9. The TRW system uses a spreader bar at the top of the array with the containment box lid and preload mechanism remaining with the storage container; the LMSC box cover and locking mechanism moves with the top of the blanket. The LMSC system must be lengthened slightly to increase its power output to the required 16.4 kW, BOL.

A summary of the key electrical characteristics of three solar-array wing options is presented in Table 3.2.1-5. The MDAC reference option is based on earlier LMSC SEP/PEP designs (2 x 4 cm cells), per decisions made at the PEP midterm review; PEP programmatics are based on this option. The LMSC option is a concept recently conceived by LMSC that employs 5.9 x 5.9 cm cells.
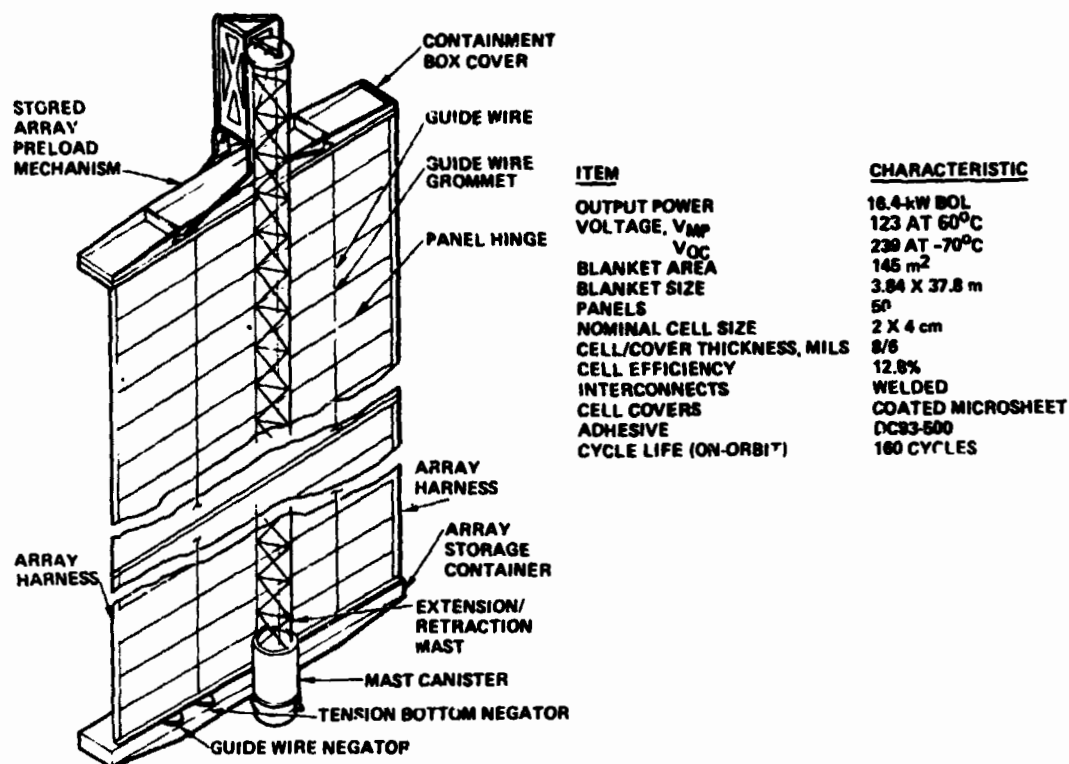
CONTAINMENT BOX COVER

STORED ARRAY PRELOAD MECHANISM

GUIDE WIRE

GUIDE WIRE GROMMET

PANEL HINGE

ARRAY HARNESS

ARRAY STORAGE CONTAINER

EXTENSION/RETRACTION MAST

MAST CANISTER

TENSION BOTTOM NEGATOR

GUIDE WIRE NEGATOR

ARRAY HARNESS

ITEM	CHARACTERISTIC
OUTPUT POWER	16.4-kW BOL
VOLTAGE, V_{MP}	123 AT 60°C
V_{OC}	239 AT –70°C
BLANKET AREA	145 m²
BLANKET SIZE	3.84 X 37.8 m
PANELS	50
NOMINAL CELL SIZE	2 X 4 cm
CELL/COVER THICKNESS, MILS	8/6
CELL EFFICIENCY	12.8%
INTERCONNECTS	WELDED
CELL COVERS	COATED MICROSHEET
ADHESIVE	DC-93-500
CYCLE LIFE (ON-ORBIT)	160 CYCLES

Figure 3.2.1-7 Solar Array Wing Characteristics (MDAC Reference)

(SINGLE CELL FOR CLARITY)

0.006-IN. COATED MICROSHEET COVERSLIDE

DC-93-500 ADHESIVE

0.008-IN. HIGH-EFFICIENCY HYBRID WRAPAROUND SOLAR CELL

FLEXIBLE PRINTED CIRCUIT SUBSTRATE

COVER SLIDE

SOLAR CELL

WELD JOINT (4 PLACES)

1/2-MIL KAPTON H-FILM/1/2 MIL ADH

FLEXIBLE SUBSTRATE WITH INTEGRAL PRINTED CIRCUIT INTECONNECTS

DC-93-500

COPPER CIRCUIT 1-1/2 MIL COPPER

1/2-MIL KAPTON H-FILM/1/2 MIL ADH

BLANKET CROSS-SECTION AT SOLAR CELL

Figure 3.2.1-8. Cross-Section and Exploded View of Solar Array Blanket

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LMSC   TRW

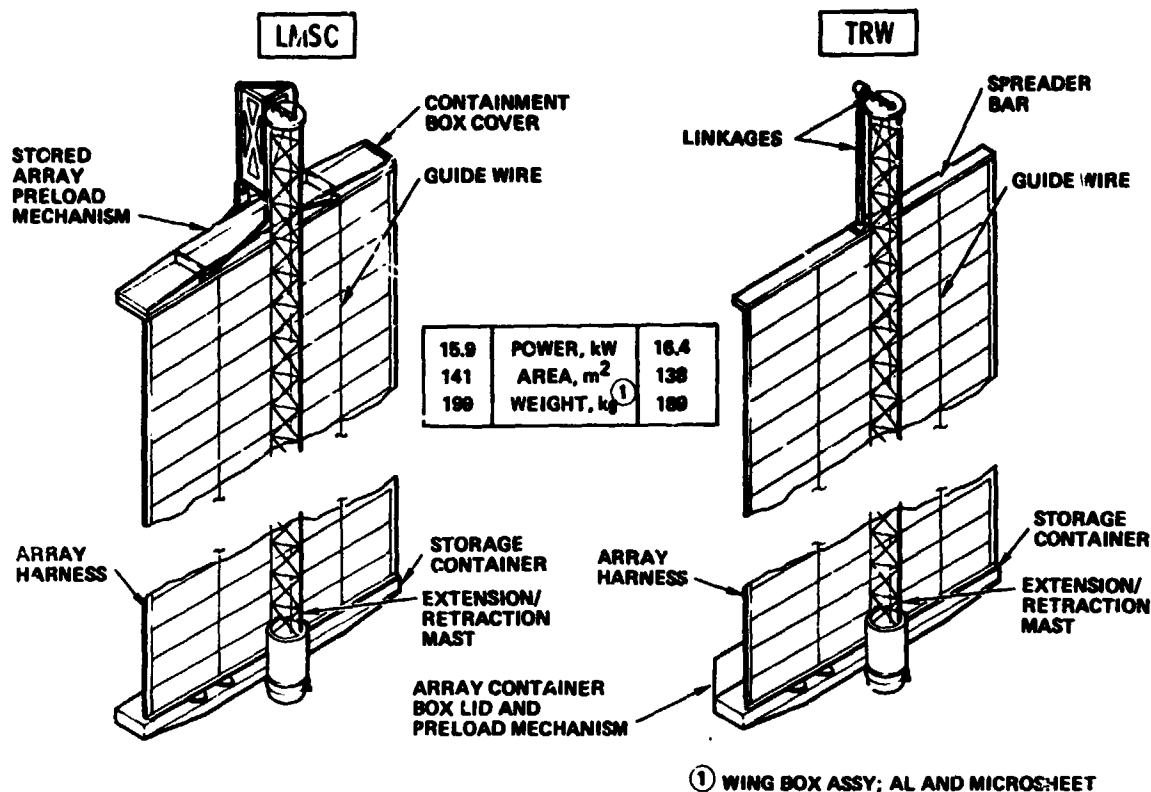| 15.9 | POWER, kW | 16.4 |
| 141 | AREA, m² | 138 |
| 199 | WEIGHT, kg ① | 189 |

① WING BOX ASSY; AL AND MICROSHEET

Figure 3.2.1-9. Solar Array Wing Options

The TRW option employs a higher efficiency solar cell (BOL), although the weight saving probably does not warrant the additional cost and the cell degrades more rapidly in the space radiation environment. A similar weight-cost trade option exists for microsheet and fused-silica covers, as may be seen on Figure 3.2.1-10. The soldered conventional interconnect offers mature technology, lower cell cost, and inspectability and repairability at the expense of blanket assembly cost and higher allowable operating temperature (for shadowing hot spots).

Figure 3.2.1-11 summarizes the key solar array performance and cost drivers that require more detailed evaluation by the solar array suppliers and the PEP contractor in future PEP efforts.

The performance of a typical solar array building-block panel (an earlier and slightly larger version of the final PEP panel) is presented in Figure 3.2.1-12. The power output is a function of panel temperature.

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Table 3.2.1-5. Characteristics of Solar Array Wing Options

| Item | Characteristics ② | | |
	MDAC Reference①	TRW	LMSC
Power output (BOL), kW	16.4	16.4	15.9
Solar cell size, cm	2 x 4	2.2 x 4	5.9 x 5.9
Solar cell type/ efficiency ③, %	HE hybrid, BSR/12.8	Shallow jct, BSF, BSR/14	HE hybrid, BSR/12.8
Voltage ④, V_{MP}/V_{OC}	123/239	122/247 (-80°C)	137/∿265
Blanket size, m	3.84 x 37.8	3.81 x 36.3	3.86 x 36.3
Blanket area, m^2	145	138	141
Blanket weight, kg	151	146	140 FS/ 147 MS
Cell cover type/ coatings ⑤	MS/MgF	MS/MgF	FS or MS/ MgF
Interconnects	Welded wraparound	Soldered conventional	Welded wraparound
Padding/stiffening	Elastomer strip/Gr/S glass/epoxy	Integral rib	Elastomer disc/Gr/S glass/epoxy

1 Midterm decision, May 17 and 18, 1979; LMSC example
2 8-MIL cells, 6-mil covers, DC93-500 adhesive
3 HE = high efficiency; BSF = back surface field; BSR = back surface reflector
4 V_{MP} = max power voltage (60°C); V_{OC} = -70°C OC voltage
5 FS = fused silica; MS = microsheet

3.2.1.4.4.2 Solar Array Selection. During the period prior to the PEP Request for Proposal, MDAC will prepare a detailed procurement specification and Statement of Work (SOW) on a solar array wing. Preliminary copies will be furnished to potential subcontractors for comments. The final specification and SOW will then be released in an RFP with proposals due back within 60 days. An evaluation will be performed and the selected subcontractor's proposal will be included in the MDAC proposal for the total PEP system.

The detailed specification will include the following:

1. Solar array wing design requirements, including output voltage, current and power, electrical connections, dark I/V testing, construction, service life, etc.
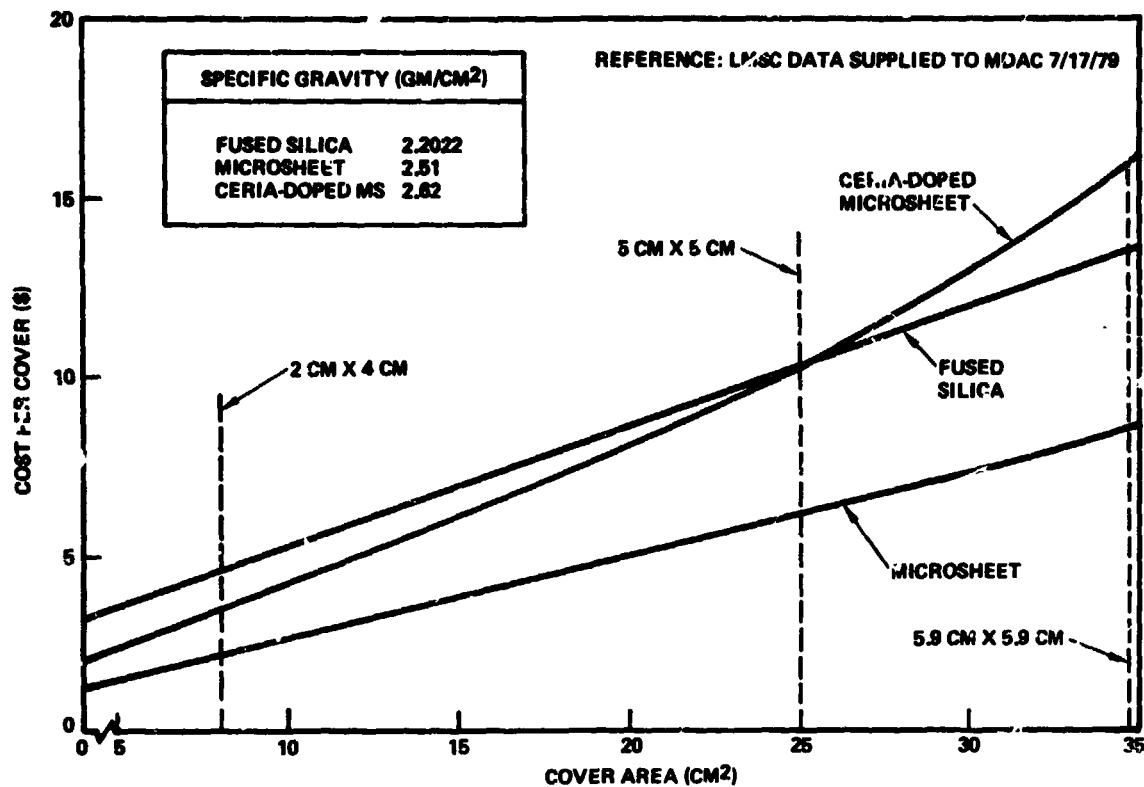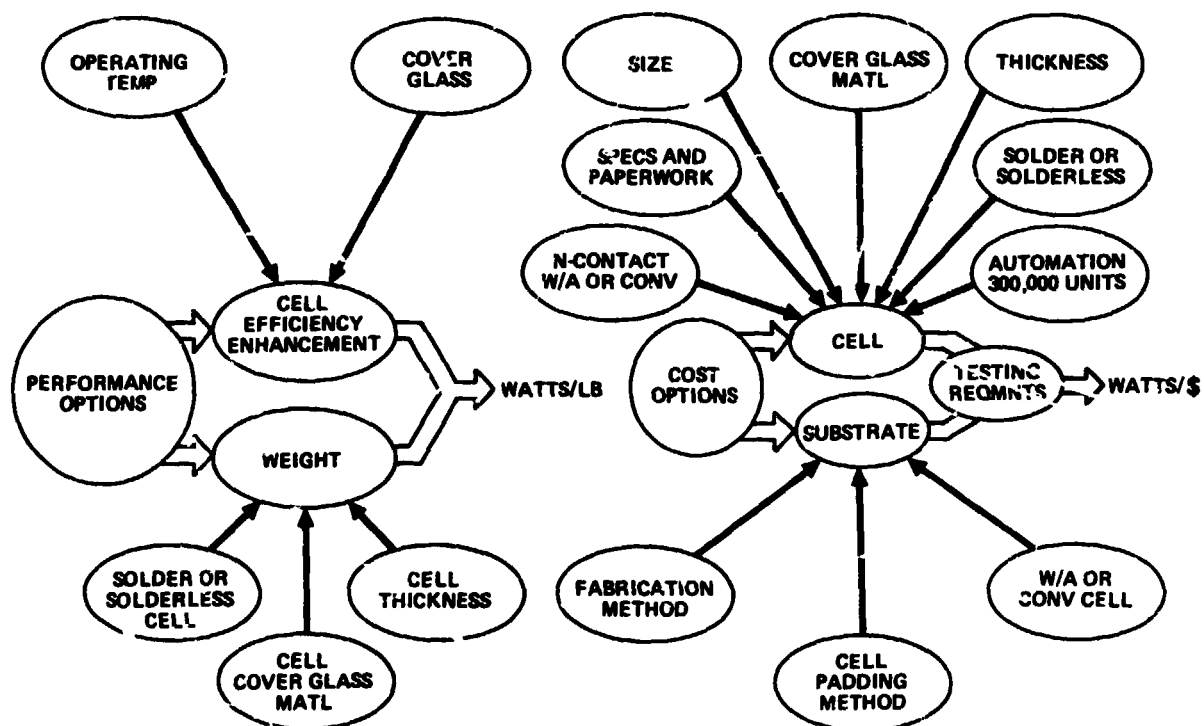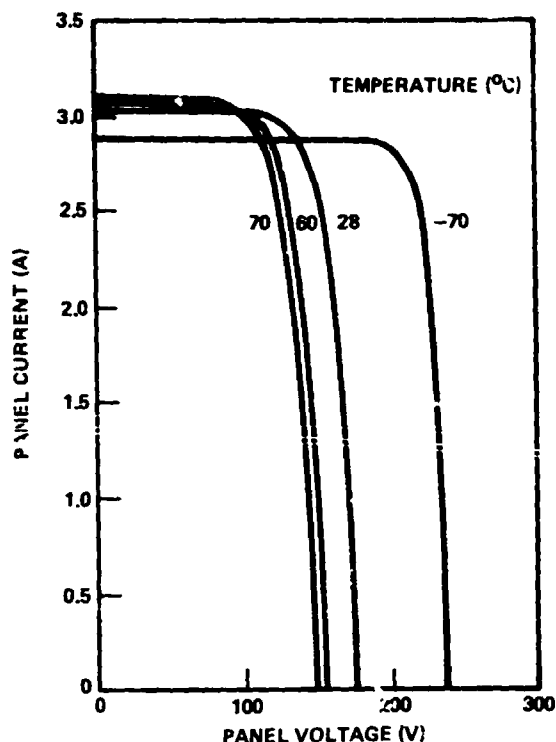
Figure 3.2.1-10. Cell Cover Cost

Figure 3.2.1-11. Solar Cell Blanket Performance and Cost Drivers

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TEMPERATURE (°C)

- ONE PANEL (0..34 X 0 m)
- 306 SERIES CELLS
- TEN STRINGS
- 325 W AT +70°C
  (2.78 A AT 117 V)
- 343 W AT +60°C
  (279 A AT 123 V)

REF: LMSC-D666410
FIGURE 3-8

Figure 3.2.1-12. Typical Solar Array Panel Characteristics

2. Blanket design requirements, including total voltage, current and power, modularity/interface, shadowing, instrumentation, size/area, weight, flatness, life, storage, tension, motion, plume loads, etc.

3. Solar cell design requirements, including interconnection, performance matching, construction, coating, bonding, etc.

4. Cover slide design requirements, including coating, transmission characteristics, bonding, etc.

5. Solar array storage box design requirements, including envelope, assembly, latching system, structural loads, interfaces, etc.

6. Deployment mast and canister design requirements, including mast strength, mast configuration constraints, deployed length, rate and power, dynamic responses, life, etc.

7. Mast and canister mechanism design requirements.

8. Electrical, mechanical, and structural interfaces.

9. Natural and induced environments.

10. Quality assurance provisions, including development tests, qualification tests, and acceptance tests.

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3.2.1.4.5 Diode Assembly Boxes (Two Required)

The diode assembly boxes contain the blocking diodes required to electrically isolate and terminate each solar array series string. Each box accepts one-half of the series strings from each solar array wing. The strings are bused together at the output (cathode) side of the diodes to form paralleled groups as required to match bus load demands.

The diodes are packaged in modules which meet Orbiter environmental specifications. Each module contains five diodes. The output of each diode is connected to a junction module for paralleling with other diodes to form a power bus. Three power buses are provided, each dedicated to supply a separate voltage regulator. Provision is also made to reconnect a number of diodes from one bus to another. In addition, interconnect wiring is provided between the two boxes and terminated in junction modules which can accept diode circuits from one box for connection to designated buses in the other box. This feature provides the capability to allocate prior to flight at least 10% of the nominal power of any one bus to any other bus in either box. The power capability of any single bus is limited, however, by its downstream voltage regulator to 10% above the nominal rating.

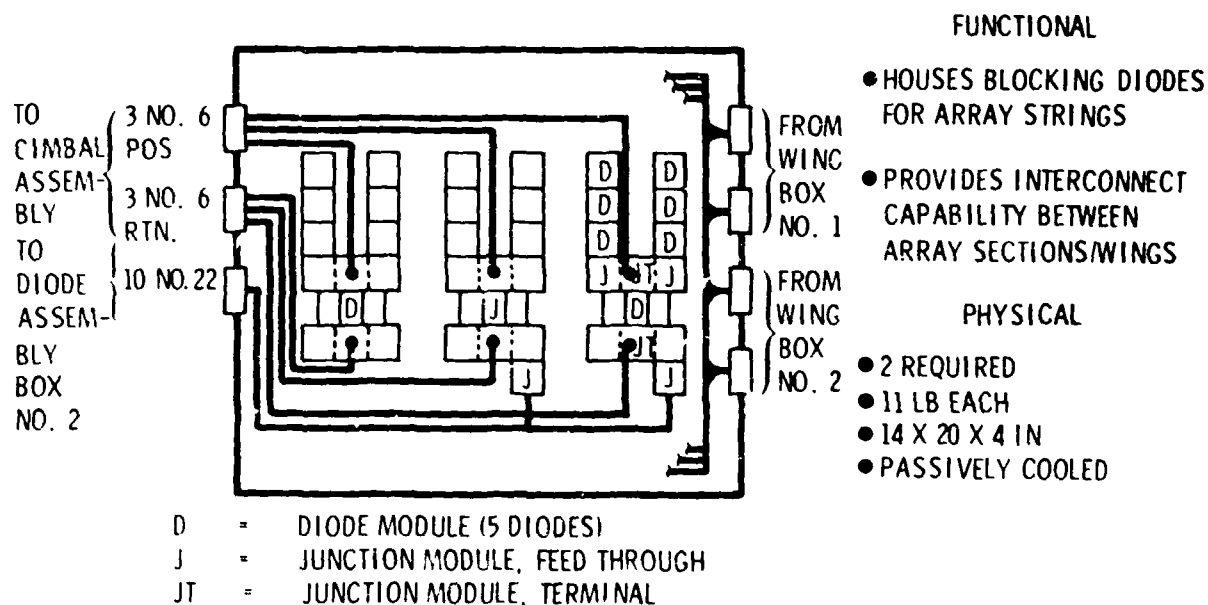The general arrangement of modules in the diode assembly box is indicated in Figure 3.2.1-13.

FUNCTIONAL

- HOUSES BLOCKING DIODES FOR ARRAY STRINGS

- PROVIDES INTERCONNECT CAPABILITY BETWEEN ARRAY SECTIONS/WINGS

PHYSICAL

- 2 REQUIRED
- 11 LB EACH
- 14 X 20 X 4 IN
- PASSIVELY COOLED

D   =   DIODE MODULE (5 DIODES)
J   =   JUNCTION MODULE, FEED THROUGH
JT =   JUNCTION MODULE, TERMINAL

Figure 3.2.1-13. Diode Assembly Box

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3.2.1.4.6 Voltage Regulators (Six Required)

The PEP voltage regulators convert raw solar array voltage to regulated
utilization voltage. In addition, they perform the important function of
controlling fuel cell output to idle power levels by remotely sensing and
closely regulating the voltage at the fuel cell terminals. Each regulator is
supplied from a dedicated (isolated) section of the solar array. The
regulators are paralleled in groups of two in the power di   ion box. Each
paralleled pair supplies one of the the three isolated buses interfacing with
the Orbiter EPS. Pulse-width modulation (PWM) control is used to regulate
output voltage. Each regulator in the flight configuration incorporates a
microprocessor which performs peak power tracking and selective protection
and control functions. Under normal operating conditions (array power
capability greater than bus load demand), the regulators share load equally
to regulate the bus voltage. When load demand exceeds array capability and
bus voltage starts to drop, both regulators operate in the peak power
tracking mode to deliver maximum available array power to the bus. The
difference between load demand and array capability is made p by the fuel
cell operating in parallel with the regulators.

If one regulator fails, the remaining unit operates autonomously. It will
regulate the bus up to its maximum current capability or the power capability
of its dedicated array section, whichever occurs first.

The regulators incorporate circuitry to detect sustained overvoltage and
overloads and utilize their microprocessor to relieve these conditions
in a preprogrammed controlled manner. The arrangement and location of
components in the flight regulators is optimized for maximum heat transfer
to the base structure. The base structure is cold-plate mounted to the
Orbiter coolant loop for efficient heat removal. Regulator dimensions are
20 x 18 x 5 in.; each regulator weighs 68 lb (30.9 kg).

3.2.1.4.7 Power-Distribution Box

The PDB controls the distribution of power from the PEP voltage regulators
to Orbiter main distribution assemblies MDA1, MDA2, and MDA3. It also
accepts the PEP system activation command from Orbiter hardwire interface
circuits and turns on the PEP MDM assembly by applying 28-volt input power.

The PDB contains three isolated power buses, each supplied by two PEP voltage
regulators. Power contactors are provided to switch each regulator on or off
the bus. Power connectors with inserts for four 0-gage conductors are
provided for the power circuits to each Orbiter MDA. Additional connectors
are provided for the six regulator remote voltage-sensing circuits, Orbiter
and PEP MDM circuits, dual redundant circuits to the dual redundant (initiators)
for the two PEP power cable in-flight disconnects at the base of the RMS,
control and power circuits for the three shunt regulators, and test functions.

The PDB also contains current monitors, relays, hybrid relays, and fuses for
required instrumentation, control, and protection functions. Cooling is by
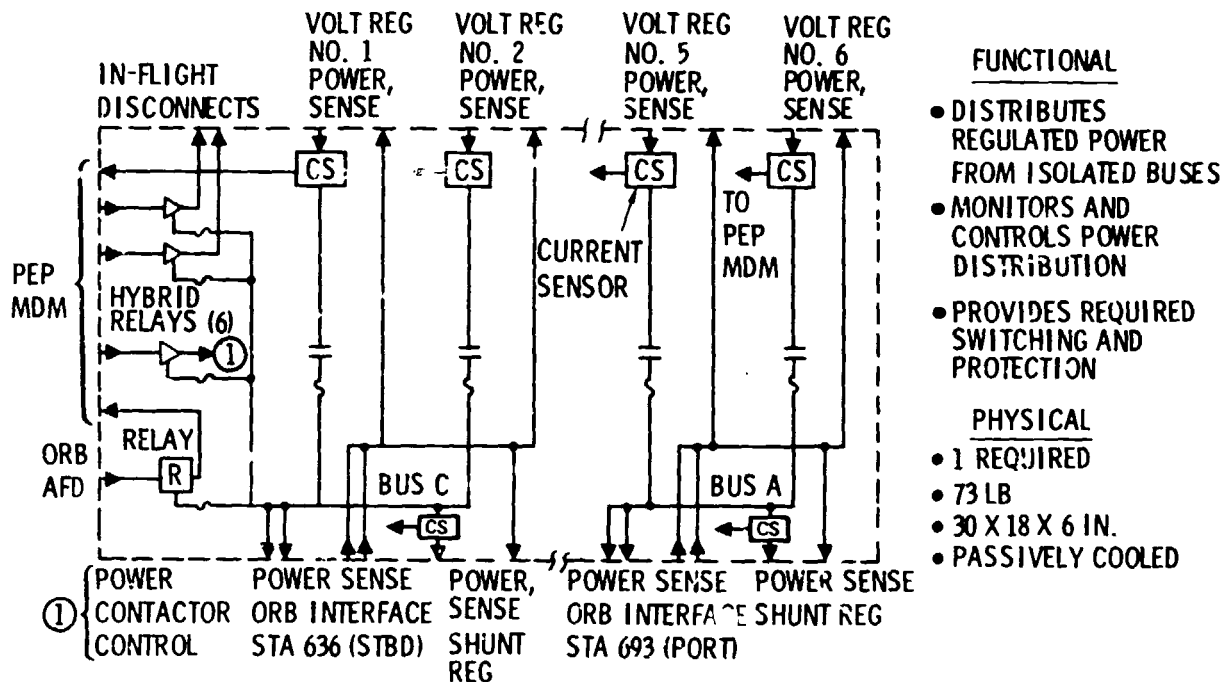given in Figure 3.2.1-14 (logic circuitry for diagnostics and control is not
shown).

Figure 3.2.1-14. Power Distribution Box

3.2.1.4.8 Shunt Regulators (Three Required)

The shunt regulators, sometimes referred to as shunt limiters or simply
shunts, provide backup overvoltage protection for the Orbit buses.* One
shunt regulator is connected to each of the three buses in the PDB. Normally,
the regulators are dormant. If a voltage above a preset threshold is sensed
at an Orbiter bus, the appropriate shunt will be enabled to turn on and draw
sufficient current to keep the voltage within allowable limits.

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The shunt regulator is designed to operate at the maximum steady-state current it could see under worst-case fault conditions for several hundred milliseconds. During this time, corrective action will be taken automatically by the protection system to clear the fault and return the shunt to its dormant state. While the shunt is active, heat is generated by the flow of current through the control element (power transistor) and load resistors. The chassis is designed to absorb the transient heat load with acceptable temperature rise; no active cooling is required.

Shunt regulator characteristics are summarized in Table 3.2.1-6.

Table 3.2.1-6. Shunt Regulator Characteristics

Functional	Physical
Normally dormant	Three required
Provides backup overvoltage protection	8 lb each
Limited operating time	9 x 9 x 5 in.
Monitored for possible failure-mode turn-on	Hardwired to PDB buses (no fuses)
	Active cooling not required

3.2.1.4.9 RMS Harness

Figure 3.2.1-15 (lower half) depicts routing of the 12 power cables from the diode assembly boxes, across the gimbal slip rings and brushes, to the umbilical connectors interfacing with the RMS.

The umbilical connectors are actuated by engage/release commands from the PEP MDM. These commands, and their talkback circuits, are carried by SPEE wiring and go through the ADA/RMS interface connectors on the grapple fixture. The umbilical connector engage (mate) command is transmitted after the SPEE/grapple fixture mechanical/electrical connections are made and the release (demate) command is sent before the SPEE/grapple fixture connections are broken.

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*Primary protection against bus overvoltage is provided by the voltage regulators, i.e., by the use of fuses and software (microprocessor) controlled internal overvoltage protection circuits.
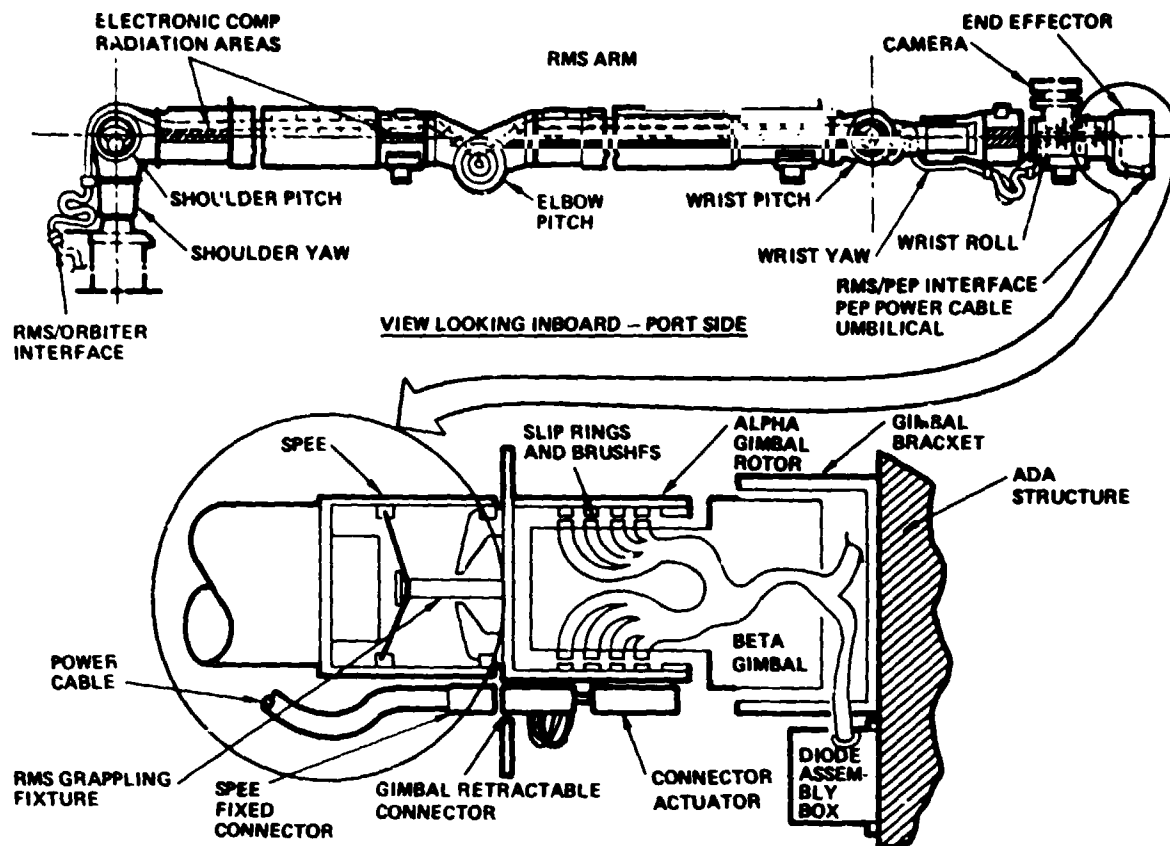
Figure 3.2.1-15. RMS Power Cable

The upper half of the figure shows routing of the harness on the outside sur-
face of the RMS, across the wrist, elbow, and shoulder joints to the two in-
flight disconnects at the RMS/Orbiter interface. The configuration of the
harness changes at specified transition points. Details of the harness design
developed by Spar Aerospace Products, Ltd., under contract to MDAC are given
in SPAR-R.940, Issue A.

The in-flight disconnects at the base of the RMS shoulder are separated on
command from the PEP MDM. Each disconnect is provided with dual redundant
nonpyrotechnic initiators. One enable/disconnect command is sent to a relay
in the power-distribution box which applies 28-volt power simultaneously to
one initiator in each disconnect. Complete separation of both connectors
should occur within 50 msec. Should separation not occur, a second command
is sent to the remaining initiators via a redundant relay in the power dis-
tribution box.

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### 3.2.2 Structural/Mechanical Subsystem

The structural/mechanical subsystem is defined as the primary structural elements that provide support and grouping of other subsystem elements and transfer flight and ground loads to the Orbiter bay structural interface. In addition, the mechanisms, both mechanical and electromechanical, which provide functions of array deployment, latching, suspension, tensioning, gimbaling, etc. are part of the structural/mechanical subsystem. The establishment of the overall PEP arrangement/configuration is closely interrelated to primary structure design; therefore, configuration options are reported in this section. Detail requirements for all subsystems are contained in the PEF System Specification, Volume 4.

3.2.2.1 Structural/Mechanical Subsysetm Requirements

A. Primary Structure. The primary structure will arrange and support other subsystem elements and provide an arrangement that can be stowed in the Orbiter bay between the airlock and the Spacelab and over the short tunnel and provide lateral clearance for both RMS arms. The dynamic envelope will be per Figure 3.2.2.1-1. The arrangement provides for mounting the ADA over Spacelab pallets. The PEP is compatible on all pallet Spacelab configuration and allows mounting of the ALA at any fore and aft bay location compatible with RMS reach. The structure will be designed to be as light as possible consistent with an adequate fatigue life and a minimum factor of safety of 1.4.

The deployable structural element will be supported on three trunnions which interface with the deployment latches that are mounted on the bridge fittings. The fixed assembly will be mounted on three trunnions which interface with fixed journals on the bridge fittings. The relative motion between the Orbiter bay walls and the PEP assemblies will be absorbed through sliding of the PEP trunnions in the structure. The bridge fittings will interface with the Orbiter stand bridge fitting attach points.

B. Gimbal Assembly. The gimbal assembly provides two-axis tracking for the deployed solar array. The gimbal will be installed on the ADA structure and interface with the RMS SPEE. The Alpha axis will be capable of continuous rotation. The Beta axis will have 90-deg minimum travel. The gimbal assembly will contain a slip ring assembly for power and signal transfer across the rotating interface.

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Figure 3.2.2.1-1. PEP System Stowed Maximum Dynamic Envelope
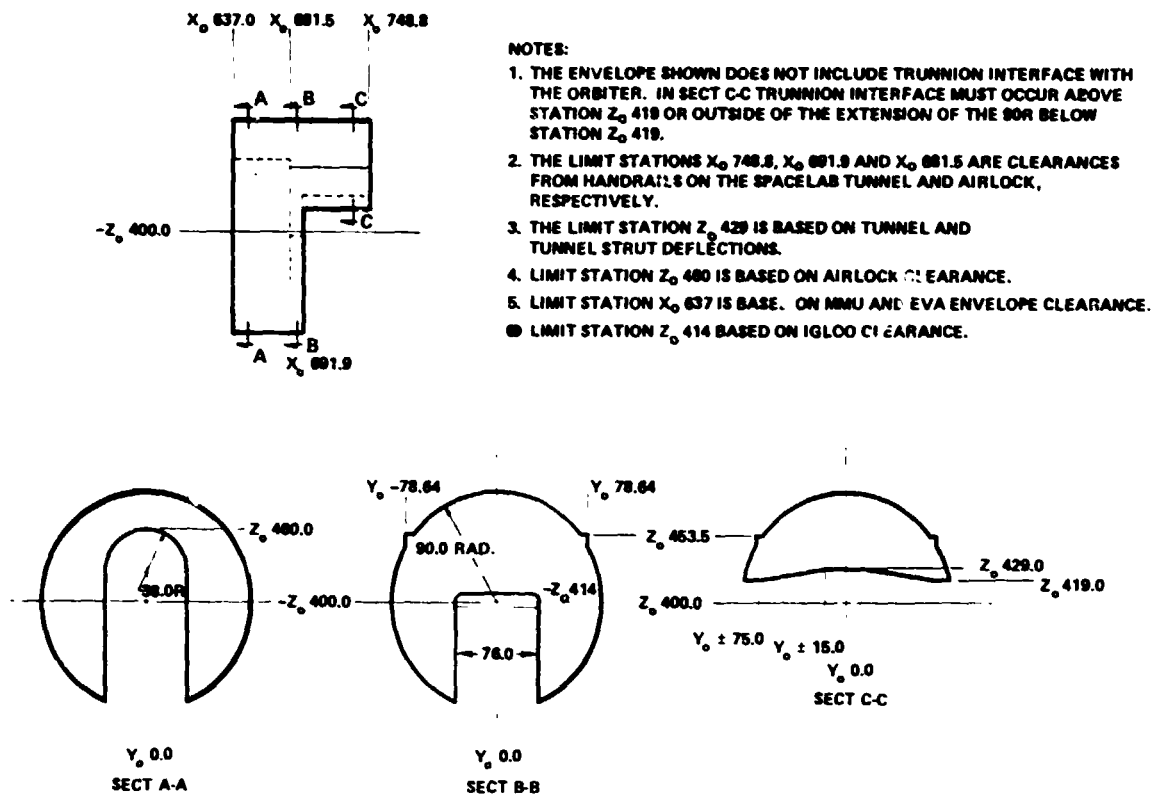
C. Mast Assembly. The mast assembly provides the extension and retraction power for the solar array blanket as well as the structural support of the blanket in the deployed position. The mast drive actuator will have redundant motors and deploy or retract the array in less than 6 min. A manual override that can be operated by an EVA astronaut will be provided. The power consumption of the actuator will be less than 100 watts. The structural strength of the mast when deployed will be sufficient to withstand VRCS plume loads activity on the solar array during attitude-hold mode and limited maneuvers.

D. Mast Canister Suspension and Lockout Mechanism. The canister suspension and lockout mechanism provides a spring suspension for the deployed solar array wing. The spring suspension will lower the natural frequency of the array both in plane and perpendicular to the array plane and limit the bending moment in the mast and the back-drive torque on the RMS joints. The suspension mechanism will incorporate a lockout device when the mast is stowed.

E. Wing Box Assembly. The wing box assembly provides the containment for the stowed array blanket. It will also contain the blanket tension and the guide wire system for the deployed array. The tension in the deployed blanket

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will be sufficient to prevent blanket contact with the support mast during dynamic excitation of the blanket. The mechanism for compressing the array blanket and locking the box cover will be part of the wing box assembly.

   F. Payload Retention Latch. The payload retention latch will be a dual three-phase AC motor, remotely operated, mechanism which will support the array deployment during launch and recovery, and will upon command release the unit for deployment on orbit. The latch will interface with the rail of a standard Orbiter bridge fitting.

3.2.2.2 Structural/Mechanical Subsystem Performance

PEP is ground-ruled for installation essentially anywhere in the Orbiter cargo bay; however, its forecast principal location will be between the airlock and the Spacelab module above the tunnel on Spacelab missions. This position is estimated to provide the most constraining geometry and the highest loads and was used for the PEP design case.

The loads criteria is based on the data presented in "Shuttle Orbiter/Cargo standards Interfaces," ICD 2-19001, Revision F, dated 22 September 1978. A review of past Shuttle payload studies resulted in the selection of a companion natural frequency in the 10-25 Hertz range as a goal.

An in-depth analysis was not possible within the scope of this study. For simplistic analysis, and to establish weight values representative of the upper limit of conservatism, the structure was modeled as a torque box beam of plates and caps. Standard stiffening and lightening practices would reduce these values by approximately 20 to 25%. Shear web elastic instability results in fairly low allowable working stresses and therefore greater material thickness and weight than what would be seen in a truss structure. A box beam of machined aluminum or molded graphite-epoxy would deal almost exclusively with column conditions and significantly higher average work stress allowables. The truss approach is estimated to show approximately 20 to 30% lower weight than the analyzed model.

3.2.2.2.1 Array Deployment Assembly Structure

3.2.2.2.1.1 Loads. The ADA is based on the concept of PEP providing special payload retention fittings to mount on the Orbiter bridge fittings. These retention fittings are sized for PEP's low weight and save considerable

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chargable weight when compared with the Orbiter's baseline retention fittings, which are designed for a maximum weight payload. These special retention fittings also place the ADA's trunnion location at station Z_o 420 instead of at the Orbiter's baseline station Z_o 414. This will allow the ADA to be installed over a Spacelab pallet.

For loads analyses, the ADA was modeled as a two-mass body with the array blanket, box, and integration structure as a distributed mass and the mast/canister and canister support structure as a point mass. Figure 3.2.2.2-1 shows this model. The indicated applied loads are positive sign for load orientation and the load values are 1-g levels.
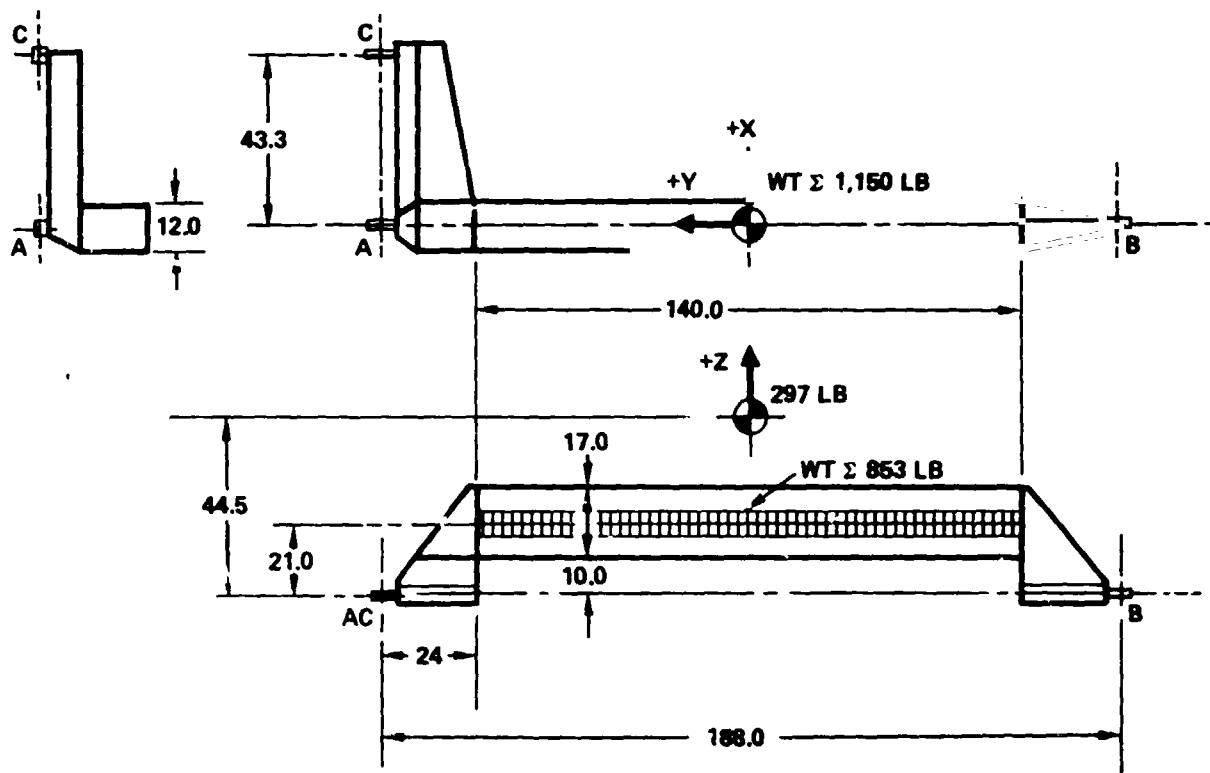

Figure 3.2.2.2-1. PEP-ADA Analytical Model

Examination of the load factor environment for Orbiter indicates that the potentially critical flight conditions are the liftoff, the terminal area energy management (TAEM) yaw maneuver, and the landing conditions. The ADA trunnion reactions were resolved for these three flight conditions and are shown in Table 3.2.2.2-1. The resulting loads, shears, bending moments, and torques on the strongback structure were derived and are shown in Table 3.2.2.2-2.

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Table 3.2.2.2-1. ADA Trunnion Reactions

Condition	Loads (lb)					
	F_{ax}	F_{ay}	F_{az}	F_{bx}	F_{bz}	F_{cz}
Liftoff	+1,840	+1.150	-697.5	+1.840	+1,271.9	+2,300.6
		-1,150	-1,028.7		+1,603.1	
TAEM yaw	-519.5	+1,437.5	+279	-517.5	-782	-647
		-1,437.5	-135		-368	
Landing	-1,035	+1,150	-955.4	-1,035	-2,580.6	-1,294
		-1,150	-1,286.6		-2,249.4	

Table 3.2.2.2-2. ADA Design Loads Summary

Condition	Loads			
	V_{max} (lb)	M_{max} (in.-lb)	Torsion (in.-lb)	Axial (lb)
Liftoff			63,725	1,150
YZ plane	1,437.5	102,152		
XY plane	1,840	125,185		
Landing			35,861	1,150
YZ plane	2,415	168,677		
XY plane	1,035	70,414		

3.2.2.2.1.2 Design. With the strongback load conditions, a design analysis was accomplished to verify the earlier weight bogies and estimates. Both aluminum and graphite-epoxy composite materials were examined.

The composite design concept was a fully webbed simplistic torque box. The final PEP reference configuration resulted in a strongback having a cross-section which is nominally 17-in. tall (Orbiter Z direction) and 12 in. wide (Orbiter X direction).

Flanges in the 12-in. width extend the overall box width to 15 in. For this geometry, the section's moments of inertia in terms of the web or wall thickness about the horizontal axis is 2,553 t and about the vertical axis, 1,512 t. The box beam was analyzed as a torque box of plates and corner caps for beam bending. No attempt was made to optimize the web design for stiffening or for lightening hole usage. The resulting web shear loads and corner cap loads for the flight conditions are shown in Table 3.2.2.2-3. For this approach, the web is critical for shear buckling at very low stress levels. Shear buckling is generally more of a concern in composites than in aluminum. Panel parameters for elastic instability in shear shows an allowable shear work stress at approximately 3,000 psi for intermediate strength graphite-epoxy, which is selected for its shear allowable. The corner caps are well-stabilized columns and a high-strength, high-modulus graphite-epoxy form was selected which gave an allowable working stress of 116,000 psi. The torque conditions are a strong contributor to the box design compared with the bending and results in a design with fairly thick webs compared with the size of the corner caps.

Table 3.2.2.2-3.   ADA Web and Cap Loads

Condition	Upper/lower shear (lb/in.)	Side shear (lb/in.)	Cap column (lb)
Liftoff	233	199	8,696
Landing	131	159	8,493

The box beam was similarly modeled in aluminum. The web elastic instability in shear resulted in working stresses of approximately 3,660 psi for the upper and lower webs and at 3,300 psi for the side webs. The corner caps were sized to their column compression loads at a working stress of 32,500 psi. Similar web thickness-to-cap area ratios occur in both the composite and aluminum materials.

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The ADA installation with the Spacelab has minimal clearances from the Spacelab module, the airlock, the tunnel, and the Orbiter RMS. The configuration was originally predicated upon clearance allocations for dynamic excursions and margins. A deflection analysis of the ADA was performed to establish the credibility of the original allocations. This analysis was performed on the assumption that dynamic data would not be available within an opportune period of time. The analysis therefore used only the quasi-steady-state load factors presented in the JSC 07700, Volume XIV companion, the "Shuttle Orbiter/Cargo Standards Interfaces," ICD 2-19001, Revision F, Change 28, dated 22 September 1978.

For purposes of the deflection analysis, the ADA model was based on some minimum thickness assumptions to ascertain the approximate upper-limit region for the deflections. Deflections were calculated to determine the contribution from:

A. Vertical and horizontal beam bending.

B. Vertical displacement of beam due to end slope rotation of the trunnion and beam-end fittings.

C. Vertical and horizontal beam deflections due to beam torque displacements.

D. Lateral translation due to beam-end fitting bending.

E. Lateral deflection due to beam-end fitting rotation about trunnion from beam-bending slope changes.

The mast canisters, in addition, see deflection contributions from the fact that they have a greater rotational arm with respect to the torque pivot center of the beam, the starboard trunnion. These deflections are shown in Table 3.2.2.2-4.

In general, the design analysis has shown that the ultimate ADA structure will exceed the original weight bogies by a small percentage and that original deflection clearance allocations are more than adequate.

3.2.2.2.2 Power Regulation and Control Assembly Structure

3.2.2.2.2.1 Loads. The configuration concept for the PRCA also uses a non-Orbiter-baseline retention fitting partly for weight optimization but largely

Table 3.2.2.2-4. ADA Deflections

Type	Amount (in.)
Vertical	
Box-beam bending – midpoint	0.526*
Box-beam displacement due to trunnion support rotation	0.131*
Box-beam corner displacement due to beam torque deflection	0.102
Box beam total	0.759
Mast/canister displacement due to beam torque deflection	0.321*
Mast/canister total	0.976*
Longitudinal	
Box-beam bending – midpoint	0.248+
Box-beam corner displacement due to beam torque deflection	0.408
Box beam total	0.656
Mast/canister displacement due to beam torque deflection	0.944+
Mast/canister total	1.192+
Lateral	
Box-beam displacement due to bending of trunnion support structure	0.122
Box-beam displacement due to trunnion support rotation from beam bending change in tip slope	0.313
Box-beam total	0.435

because it is installed adjacent to the Orbiter RMS station. Special bridge
fittings are required to provide a payload support at such a location. The
PRCA supports are at Orbiter station Z_o 408.5. This location, however, had
little effect in determining the loads on the support points and with the
PRCA structure. After the analysis was completed, it was discovered that the
Spacelab subsystems igloo for pallet-only missions could be located at the
same in-bay station and that current definition placed its upper-clearance

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requirement higher than the PRCA now accommodates. The PRCA beam would have to adopt some form of arch geometry; however, the amount should not have a significant bearing on the PRCA structural thicknesses and weight.

The PRCA was modeled as a stick beam with several point masses in determining the support reactions, shears, bending moments, and torque loads on its structure. Figure 3.2.2.2-2 shows this model. Table 3.2.2.2-5 lists the various loads for the flight conditions.
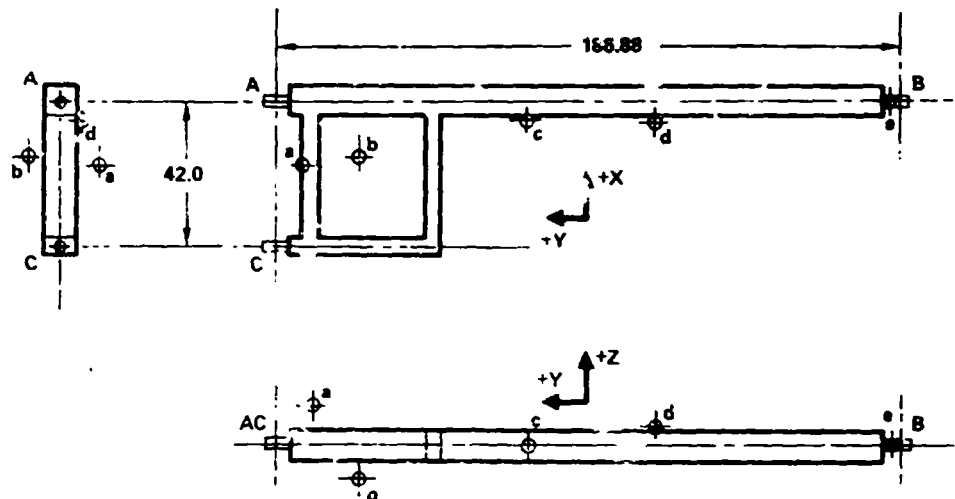
ITF	ΔX (IN)	ΔY (IN)	ΔZ (IN)	WEIGHT (LB)
a	-18.40	11.43	+11.70	94
b	-16.24	-25.23	-9.79	48?
c	-4.50	-75.94	0.0	59
d	-6.0?	-113.44	+6.5	27
e	0.0	-182.94	0.0	3

REFERENCE ORIGIN FOR ΔX, ΔZ IS A–B; FOR ΔY IS A–C

Figure 3.2.2.2-2. PEP-PRCA Analytical Model

3.2.2.2.2.2 Design. The PRCA configuration is primarily a concentrated grouping of equipment near the starboard sidewall of the Orbiter. The selected structure is a small section box beam spanning the Orbiter cargo bay width with a simple box-beam ladder section attached to carry the equipment. The bulk of the equipment would be cold-plate-mounted, with the cold plates supported on the ladder section. Design analysis similar to that performed for the ADA resulted in web shear loads and corner cap loads shown in Table 3.2.2.2-6. In graphite-epoxy materials, the elastic shear buckling allows a working stress of approximately 3,700 psi in the webs while the caps are worked at approximately 115,000 psi. The comparative analysis in aluminum

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Table 3.2.2.2-5.   PRCA Loads Summary

Trunnion loads

Condition	Loads (lb)				
	F_{ax}	F_{ay}	F_{az}	F_{bx}	F_{bz}
Liftoff	1,761.3	±664	1,375.1	365.7	285.9
Landing	990.1	±664	2,310.5	205.9	479.6

Beam maximum loads

Condition	Loads			
	V_{max} (lb)	M_{max} (in.-lb)	Torsion (in.-lb)	Axial (lb)
Liftoff			30,150	644
YZ plane	1,375	31,450		
XY plane	1,761	40,276		
Landing			33,326	664
YZ plane	2,310	52,830		
XY plane	990	22,646		

Table 3.2.2.2-6.   PRCA Beam Web and Cap Loads

Condition	Upper/lower web shear (lb/in.)	Side web shear (lb/in.)	Cap column (lb)
Liftoff	342 maximum	322 maximum	5,665
	55 average	49 average	
Landing	323 maximum	405 maximum	5,583
	44 average	69 average	

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shows approximately the same relationship to composite as was found in the ADA structure. In general, the comparison shows the webs to be working the materials at nearly the same allowables while the caps work the composite to nearly three and one-half times the allowable for aluminum. Graphite-epoxy could, therefore, expect to save an appreciable percentage of weight but at a definite increase in cost.

3.2.2.3 Major Structural Mechanical Trades and Analysis

3.2.2.3.1 PRCA Configuration

The PRCA consists of 11 major elements with a total weight of approximately 650 lb. Three configurations were evaluated; they were a three-trunnion supported beam transversely spanning the Orbiter bay, a single side mounting using a single custom-designed bridge fitting, and a thermally passive system with a heat pipe radiator transversely spanning the cargo bay and supporting the other subsystem elements on the under side. The thermally passive system was not compared to the other two options as a configuration or for structural/mechanical trade but was evaluated in combination with the thermal control system trade as discussed in Section 3.2.4.3. Figure 3.2.2.3-1 illustrates the three PRCA configurations that were evaluated.

The comparison between the single side mount and the trunnion-supported beam configuration has led to the selection of the latter as the reference configuration for the study. The weight of the single side-mount configurations would be approximately 25 lb lighter; however, without the bay spanning beam, the power cables would have to be routed below the tunnel and attached directly to C-biter structure. This condition would significantly impact PEP ground installation and removal and/or add significantly to Orbiter scar weight.

Secondary issues which were evaluated during the study produced design solutions and rationale as follows:

● The voltage regulators are mounted low with respect to the beam structure to reduce damage susceptibility of cold plates and to provide visibility from the aft flight window to the ADA deployment latch.

● The components are clustered on the right side of the Orbiter because two of the three Orbiter power bus interfaces are located on the right side.
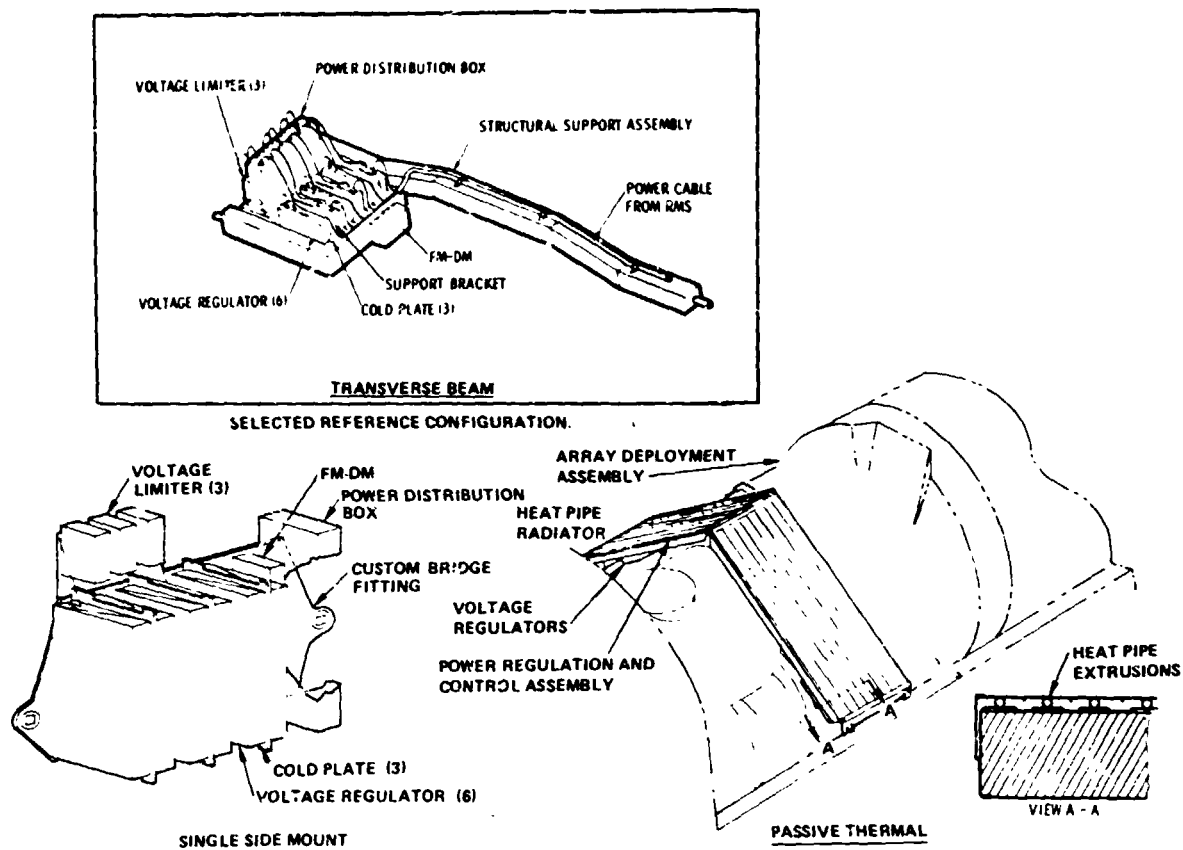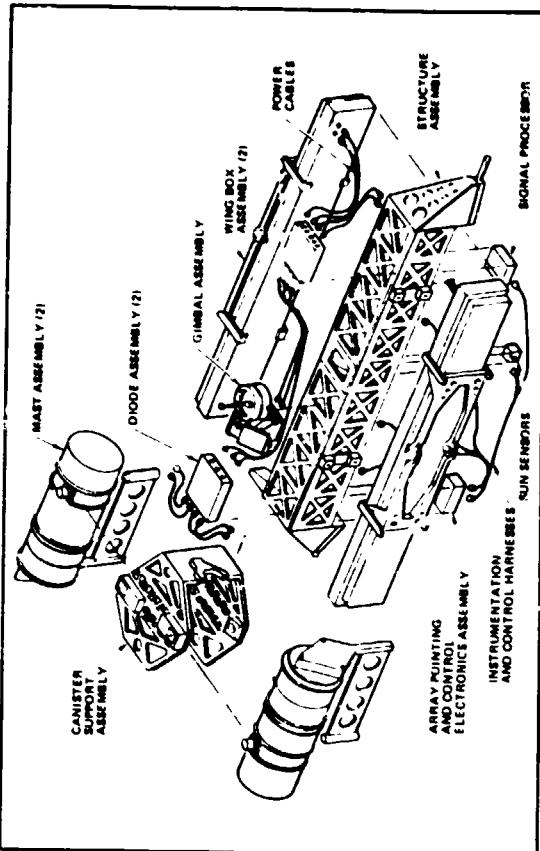
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Figure 3.2.2.3-1. Three PRCA Configurations

● The power-distribution box is mounted high on the beam to expose as much surface area as possible to space environment for passive thermal control.

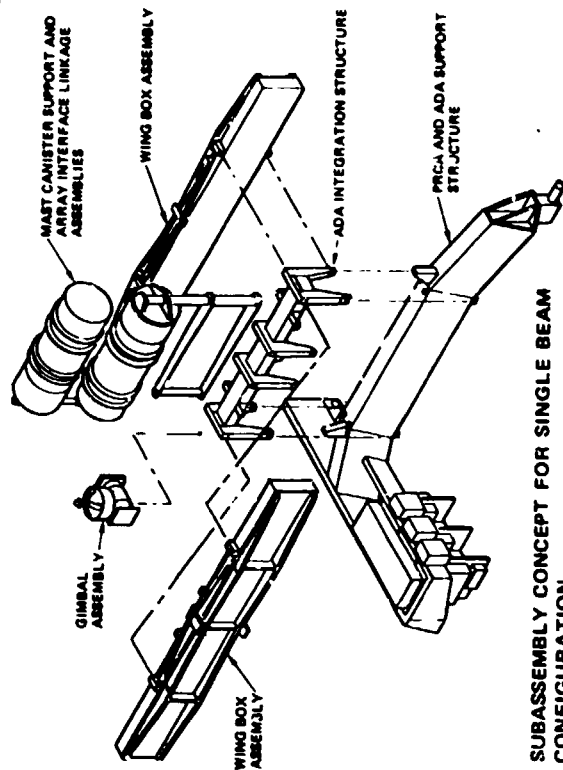3.2.2.3.2 <u>ADA Configuration</u>

The ADA consists of 12 major elements with a total weight of approximately 1,400 lb. During this phase of the study four basic configurations were evaluated. The configurations are illustrated by Figure 3.2.2.3-2 and are numbered 2 through 5. Configuration 1 is assigned to the Phase A baseline as described in MDAC Report MDC G7555.

Each configuration was evaluated using weight, cost, interface requirements, technical feasibility, aft bay mounting, and envelope penetration as the criteria. Initially weight was considered to be the primary design driver. Using this criteria, Configuration 2, the integral wing box design, was selected based on minimum structural weight. Subsequently, it was determined that utilizing the wing box to form the side shear panels for the support beam
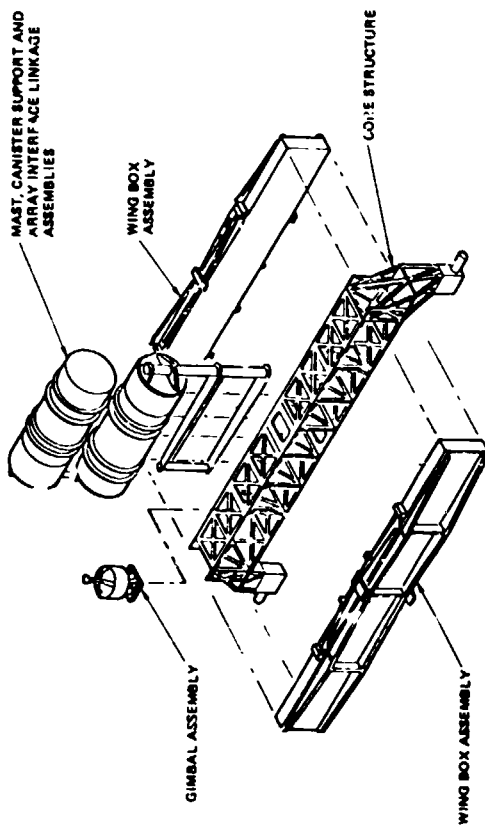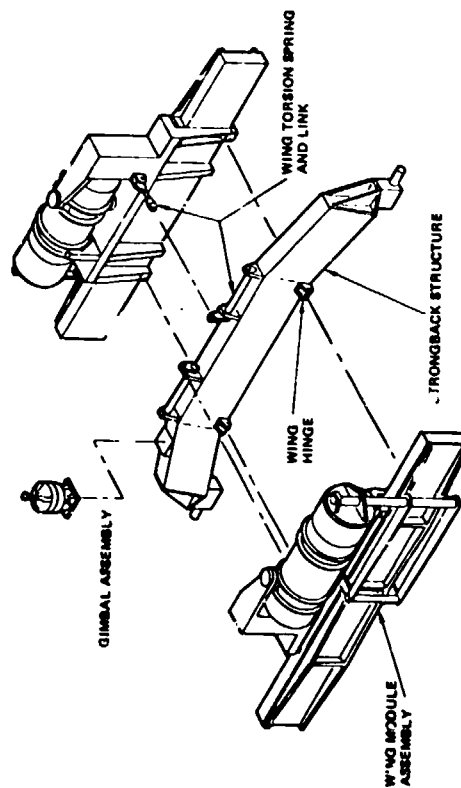
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SELECTED REFERENCE CONFIGURATION

MAST ASSEMBLY (2)

DIODE ASSEMBLY (2)

POWER CABLES

WING BOX ASSEMBLY (2)

GIMBAL ASSEMBLY

STRUCTURE ASSEMBLY

SIGNAL PROCESSOR

SUN SENSORS

CANISTER SUPPORT ASSEMBLY

ARRAY POINTING AND CONTROL ELECTRONICS ASSEMBLY

INSTRUMENTATION AND CONTROL HARNESSES

SUBASSEMBLY CONCEPT FOR STRONGBACK CONFIGURATION

MAST, CANISTER SUPPORT AND ARRAY INTERFACE LINKAGE ASSEMBLIES

WING BOX ASSEMBLY

ADA INTEGRATION STRUCTURE

PRGA AND ADA SUPPORT STRUCTURE

GIMBAL ASSEMBLY

WING BOX ASSEMBLY

SUBASSEMBLY CONCEPT FOR SINGLE BEAM CONFIGURATION

MAST, CANISTER SUPPORT AND ARRAY INTERFACE LINKAGE ASSEMBLIES

WING BOX ASSEMBLY

CORE STRUCTURE

GIMBAL ASSEMBLY

WING BOX ASSEMBLY

SUBASSEMBLY CONCEPT FOR INTEGRAL WING BOX CONFIGURATION

WING TORSION SPRING AND LINK

STRONGBACK STRUCTURE

WING HINGE

GIMBAL ASSEMBLY

WING MODULE ASSEMBLY

SUBASSEMBLY CONCEPT FOR MODULAR WING CONFIGURATION

Figure 3.2.2.3-2. PEP ADA Configuration Options Evaluated

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would result in an unacceptable interface between the array wing box and the support structure. Deflections of the loaded wing box caused by compression of the blanket would make a close-tolerance multiattachment shear connection difficult to accomplish. Because of the interface problem, Configuration 3, the strongback design, was selected as the reference configuration for the study.

Configuration 3 is approximately 24 lb heavier than Configuration 2; however, it weighs less than Configurations 4 and 5. A sixth configuration, the fixed-canister design, is a variation which could be applied to any of the support structure configurations. Configuration 4, the modular wing design, has the advantage of modular assembly and adjustment of the wing box and mast canister. The disadvantage is a considerable increase in weight because of the structural redundancy and the envelope restrictions requiring high loads to be carried through small structural cross-sections. Configuration 5 combines the structural support of the ADA and the PRCA equipment. This arrangement is slightly heavier than the reference configuration, but the major objection to the arrangement is the inability to stow the package at any location within the Orbiter bay. Since the ADA and PRCA are combined, the electrical power and coolant interfaces in the Orbiter bay would require relocation if the single beam concept were moved aft. Aft location of the deployable portion of the assembly by the use of a second support beam is possible, but this would add to the structural weight of the system.

3.2.2.3.3 Rotating and Fixed-Mast Canister

Figure 3.2.2.3-3 illustrates a fixed-canister and a rotating-canister configuration. The evaluation criteria were envelope constraints, weight, mechanical complexity, and array dynamic response.

The rotating-canister configuration was selected for the referenced design on the basis that the absolute length of the mast canister was a "soft" dimension which can only be determined with a detail design of the mast, the envelope limits are more flexible with the mast canister axis transverse in the bay, and the dynamics of the deployed array with the mast off the center line have not been fully analyzed.

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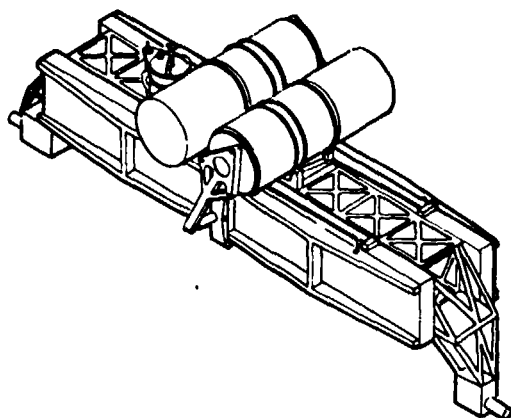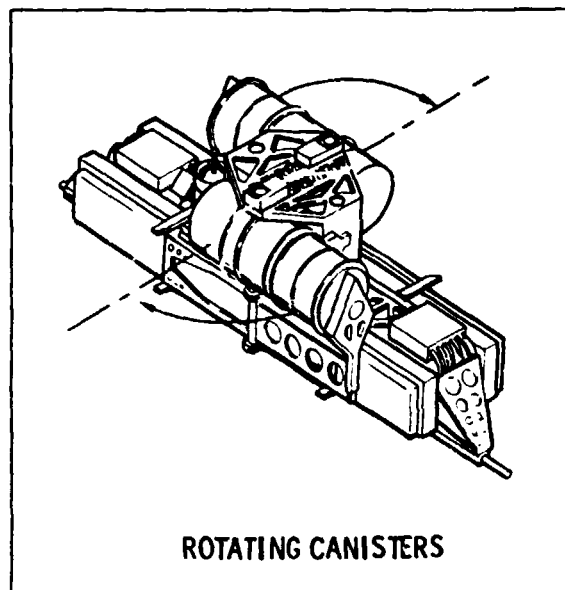
FIXED CANISTERS   ROTATING CANISTERS

                        SELECTED REFERENCE CONFIGURATION

Figure 3.2.2.3-3. ADA Mast Canister Mounting Options

The potential slight intrusion of the rotating-canister configuration into the
area outside of the 90-in. radius at Zo 475 can be tolerated because of the
Orbiter bay door/radiator contour at that location. The fore and aft limits
for canister length for the fixed-canister configuration are determined by the
structure of the airlock forward and the Spacelab aft. A canister length of
55.7 in. will allow the fixed-canister configuration to fit within the envel-
ope constraints, but additional length would preclude this arrangement.

Because of the off-center mast in the fixed-canister configuration the bending
moment in the mast caused by blanket tension increases. Figure 3.2.2.2-4 com-
pares the two configurations in terms of geometry mast size and weight. The
fixed- canister configuration would weigh the same as the rotating-canister
configuration because the mast increased weight is offset by a reduction in
rotating mechanism weight.

The dynamic analysis of the deployed array was accomplished with a symetrical
configuration. A preliminary examination of the dynamics of the off-center
mast indicates that the array dynamic responses would be satisfactory, but
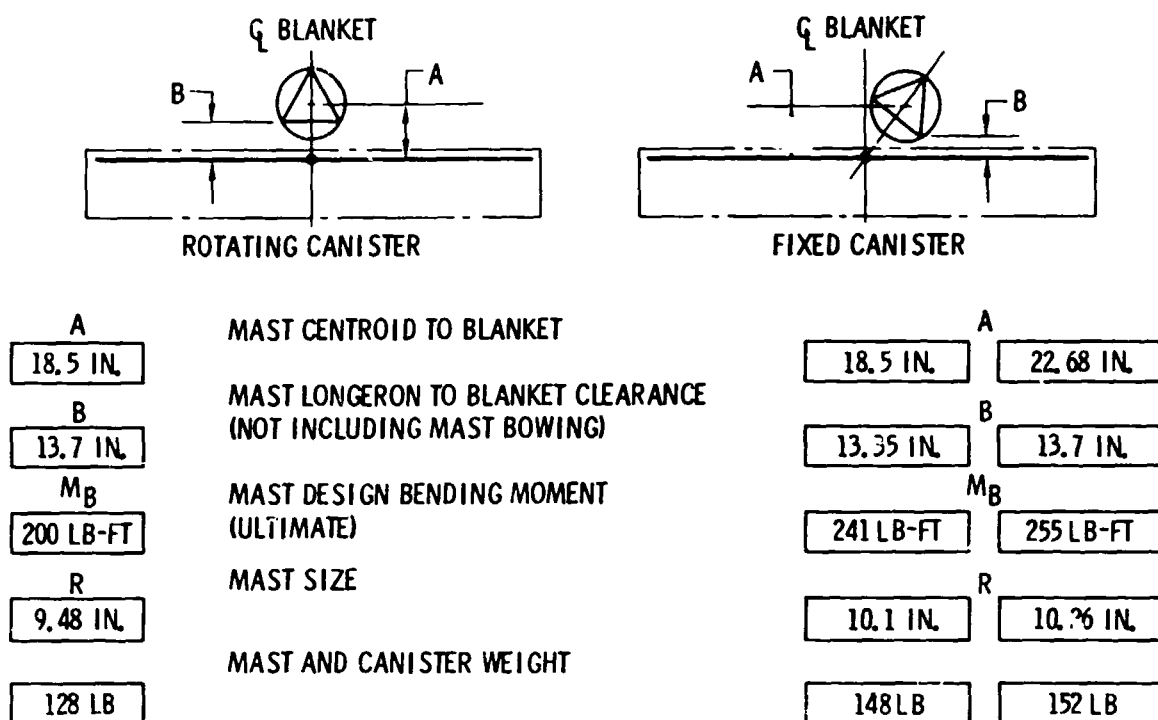more analysis is necessary to verify the preliminary conclusion.

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	ROTATING CANISTER	FIXED CANISTER	
	A 18.5 IN.	MAST CENTROID TO BLANKET	**A** 18.5 IN. / 22.68 IN.
	B 13.7 IN.	MAST LONGERON TO BLANKET CLEARANCE (NOT INCLUDING MAST BOWING)	**B** 13.35 IN. / 13.7 IN.
	M_B 200 LB-FT	MAST DESIGN BENDING MOMENT (ULTIMATE)	**M_B** 241 LB-FT / 255 LB-FT
	R 9.48 IN.	MAST SIZE	**R** 10.1 IN. / 10.76 IN.
	128 LB	MAST AND CANISTER WEIGHT	148 LB / 152 LB

Figure 3.2.2.3-4. Sizing of Fixed and Rotating Canisters

3.2.2.3.4 Array Wing Box Length Analysis

The reference configuration has the wing box mounted transversely in the
Orbiter bay between the port and starboard RMS arms. Figure 3.2.2.3-5
illustrates the array wing box superimposed on a cross-section of the Orbiter
bay. The initial PEP concepts were based on the SEP solar array technology and
used a box length of 159.04 in. Upon closer inspection, it was found that if
both RMS's were installed and the deflections of RMS, longeron, and wing box
were considered, there would be a physical interference between the RMS and
the wing box. Using the maximum Orbiter longeron deflection for any point
within the bay, a 2-deg rotational deflection of both RMS arms, a $\pm$1-in.
dynamic excursion of the wing box, and an 8.5-in. radius dynamics envelope for
the RMS, it was established that the wing box maximum length should be 152.8
in. At this length the wing box penetrates the 90-in. radius payload envelope;
however, there is no danger of physical contact with the Orbiter side of the
interface and the box was considered acceptable and was selected for the
referenced design. If the wing box were designed to be completely contained
within the 90-in. radius envelope the box length could not exceed 144 in. To

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MAST LENGTH	BOX LENGTH	NO. PANELS
120.8 FT	159 IN.	48
125.6	152.6	50
133.2	144	53

SELECTED FOR REFERENCE DESIGN

*SIDEWALL LATERAL DEFLECTIONS ARE 1.83 IN. AND 2.65 IN. MAXIMUM SIMULTANEOUS FOR A GIVEN FLIGHT CONDITION AND AT THE WORST LOCATION ($\sim X_o$ 903).

**THIS ANGULAR DEFLECTION IS DUE TO LONGERON ROLL BASED ON AN ASSUMPTION OF 1.0 DEGREE EACH FOR SIDEWALL BENDING AND LOCAL LOADING.
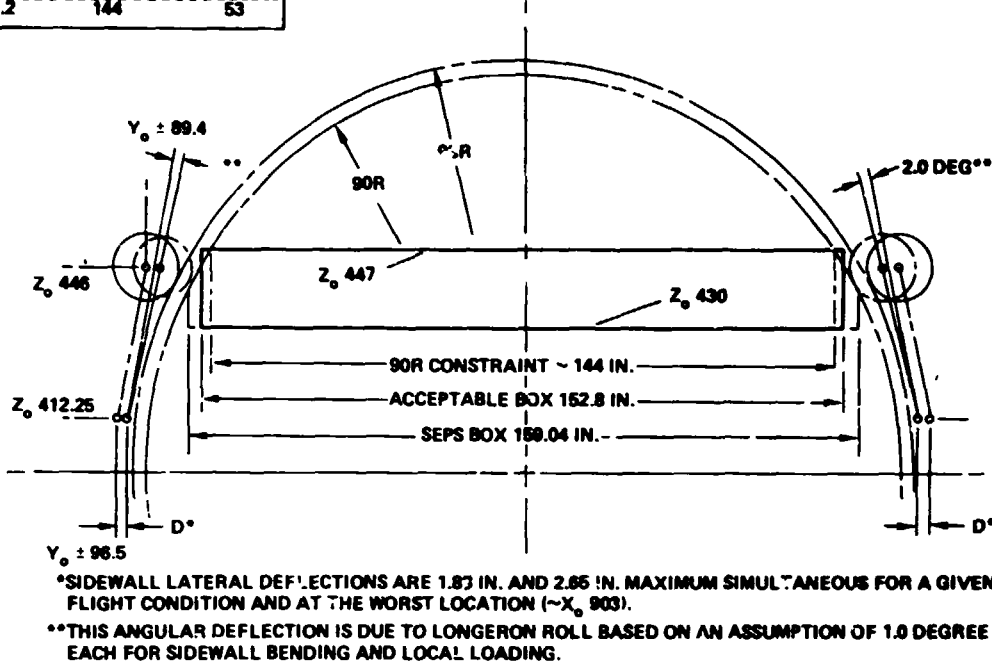
Figure 3.2.2.3-5. Wing Box Length Constraints and Deflections

maintain the area of the array blanket constant as the box length is reduced, the blanket and the mast length must , increased. Figure 3.2.2.3-5 indicates mast length as a function of box length.

3.2.2.3.5 Aluminum vs Composite Structure

Structure design layouts were prepared using both aluminum and graphite-epoxy materials. If graphite-epoxy is used for the referenced design instead of aluminum a weight reduction of 104 lb can be achieved. Aluminum was selected for the referenced design because of the high cost and time required for development of the composite. The composite is a strong alternate if a weight savings cost of approximately $15,000 per pound can be justified.

3.2.2.4 Structural Mechanical Subsystem Description

Figure 3.2.2.4-1 illustrates the drawing tree for the reference PEP configuration. The shaded elements are all part of the structural mechanical subsystem. The PEP system installation is composed of three major elements: the ADA, the PRCA, and the interface kit. The ADA contains all of t' elements of the PEP system that are deployed out of the Orbiter bay during orbital
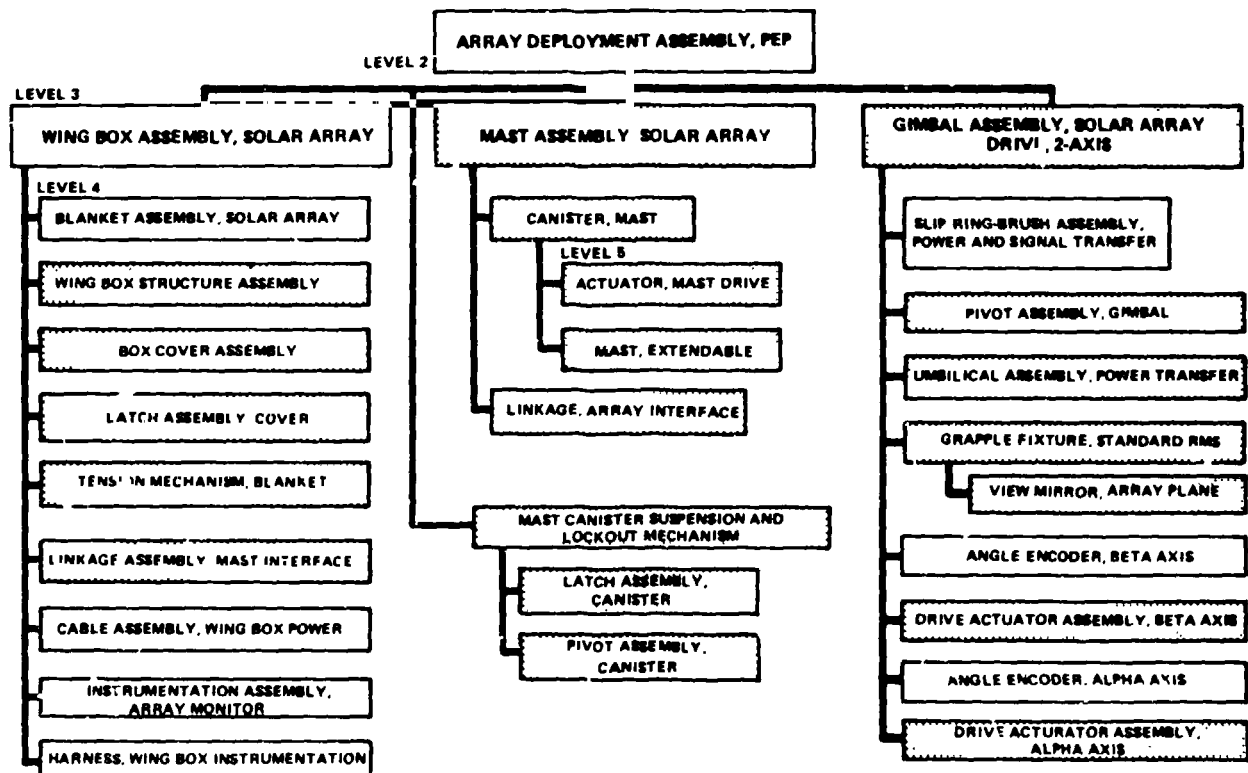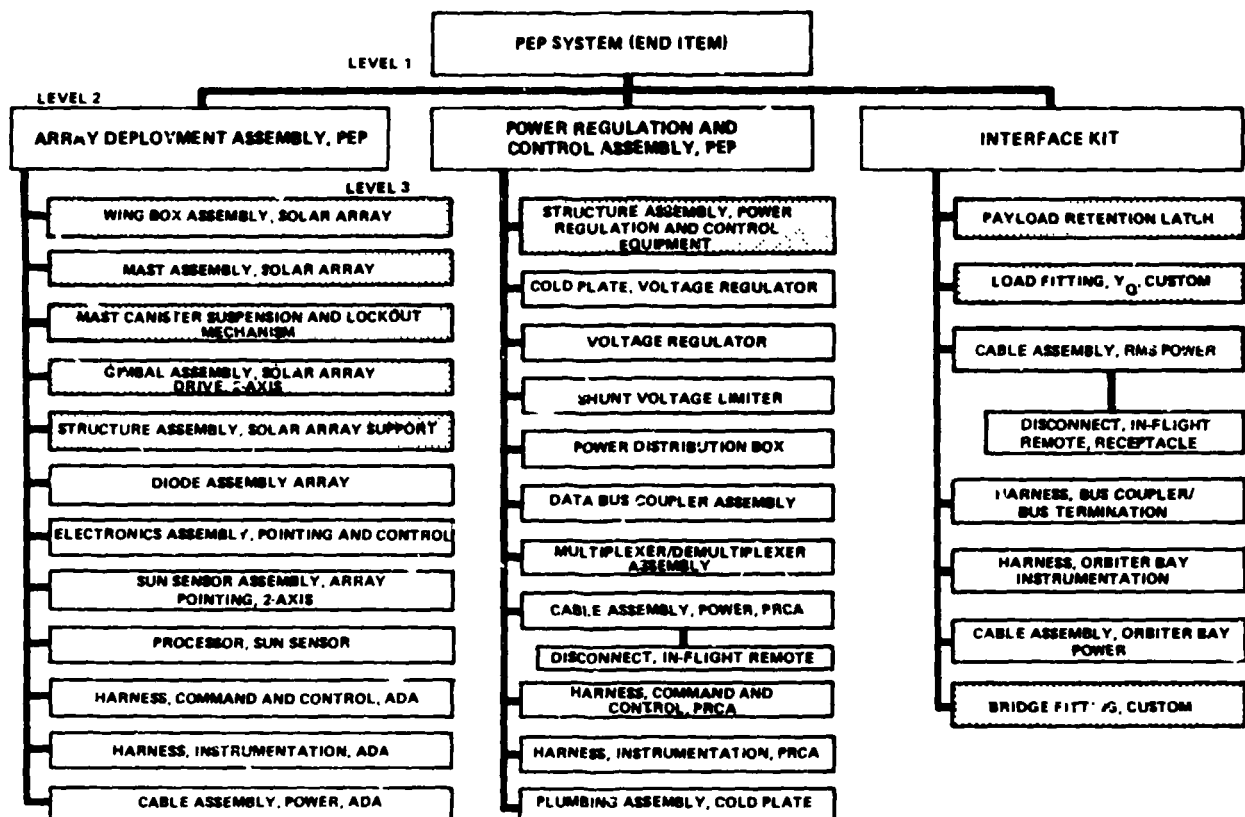
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PEP SYSTEM (END ITEM)

LEVEL 1

LEVEL 2

ARRAY DEPLOYMENT ASSEMBLY, PEP

LEVEL 3

- WING BOX ASSEMBLY, SOLAR ARRAY
- MAST ASSEMBLY, SOLAR ARRAY
- MAST CANISTER SUSPENSION AND LOCKOUT MECHANISM
- GIMBAL ASSEMBLY, SOLAR ARRAY DRIVE, 2-AXIS
- STRUCTURE ASSEMBLY, SOLAR ARRAY SUPPORT
- DIODE ASSEMBLY ARRAY
- ELECTRONICS ASSEMBLY, POINTING AND CONTROL
- SUN SENSOR ASSEMBLY, ARRAY POINTING, 2-AXIS
- PROCESSOR, SUN SENSOR
- HARNESS, COMMAND AND CONTROL, ADA
- HARNESS, INSTRUMENTATION, ADA
- CABLE ASSEMBLY, POWER, ADA

POWER REGULATION AND CONTROL ASSEMBLY, PEP

- STRUCTURE ASSEMBLY, POWER REGULATION AND CONTROL EQUIPMENT
- COLD PLATE, VOLTAGE REGULATOR
- VOLTAGE REGULATOR
- SHUNT VOLTAGE LIMITER
- POWER DISTRIBUTION BOX
- DATA BUS COUPLER ASSEMBLY
- MULTIPLEXER/DEMULTIPLEXER ASSEMBLY
- CABLE ASSEMBLY, POWER, PRCA
  - DISCONNECT, IN-FLIGHT REMOTE
- HARNESS, COMMAND AND CONTROL, PRCA
- HARNESS, INSTRUMENTATION, PRCA
- PLUMBING ASSEMBLY, COLD PLATE

INTERFACE KIT

- PAYLOAD RETENTION LATCH
- LOAD FITTING, Y_O CUSTOM
- CABLE ASSEMBLY, RMS POWER
  - DISCONNECT, IN-FLIGHT REMOTE, RECEPTACLE
- HARNESS, BUS COUPLER/ BUS TERMINATION
- HARNESS, ORBITER BAY INSTRUMENTATION
- CABLE ASSEMBLY, ORBITER BAY POWER
- BRIDGE FITTING, CUSTOM

ARRAY DEPLOYMENT ASSEMBLY, PEP

LEVEL 2

LEVEL 3

WING BOX ASSEMBLY, SOLAR ARRAY

LEVEL 4

- BLANKET ASSEMBLY, SOLAR ARRAY
- WING BOX STRUCTURE ASSEMBLY
- BOX COVER ASSEMBLY
- LATCH ASSEMBLY COVER
- TENSION MECHANISM, BLANKET
- LINKAGE ASSEMBLY MAST INTERFACE
- CABLE ASSEMBLY, WING BOX POWER
- INSTRUMENTATION ASSEMBLY, ARRAY MONITOR
- HARNESS, WING BOX INSTRUMENTATION

MAST ASSEMBLY SOLAR ARRAY

- CANISTER, MAST
  LEVEL 5
  - ACTUATOR, MAST DRIVE
  - MAST, EXTENDABLE
- LINKAGE, ARRAY INTERFACE
- MAST CANISTER SUSPENSION AND LOCKOUT MECHANISM
  - LATCH ASSEMBLY, CANISTER
  - PIVOT ASSEMBLY, CANISTER

GIMBAL ASSEMBLY, SOLAR ARRAY DRIVE, 2-AXIS

- SLIP RING-BRUSH ASSEMBLY, POWER AND SIGNAL TRANSFER
- PIVOT ASSEMBLY, GIMBAL
- UMBILICAL ASSEMBLY, POWER TRANSFER
- GRAPPLE FIXTURE, STANDARD RMS
  - VIEW MIRROR, ARRAY PLANE
- ANGLE ENCODER, BETA AXIS
- DRIVE ACTUATOR ASSEMBLY, BETA AXIS
- ANGLE ENCODER, ALPHA AXIS
- DRIVE ACTUATOR ASSEMBLY, ALPHA AXIS

Figure 3.2.2.4-1. PEP Drawing Tree

operations. The PRCA contains all the elements of the system that are assem-
bled external to the Orbiter but after installation remain fixed in the
Orbiter bay during orbital operations. The installation kit contains all of
the hardware required to install the entire PEP system and provide the struc-
tural electrical and thermal interfaces with the Orbiter.

Figure 3.2.2.4-2 illustrates the PEP reference configuration as installed in
the Orbiter. The PRCA is mounted forward with the centerline of the bay
spanning support beam at Station Xo 687.5. The voltage regulators and associa-
ted equipment are mounted on the starboard side. The ADA is mounted aft of the
PRCA. The centerline of the ADA support structure is located at station Xo
715.
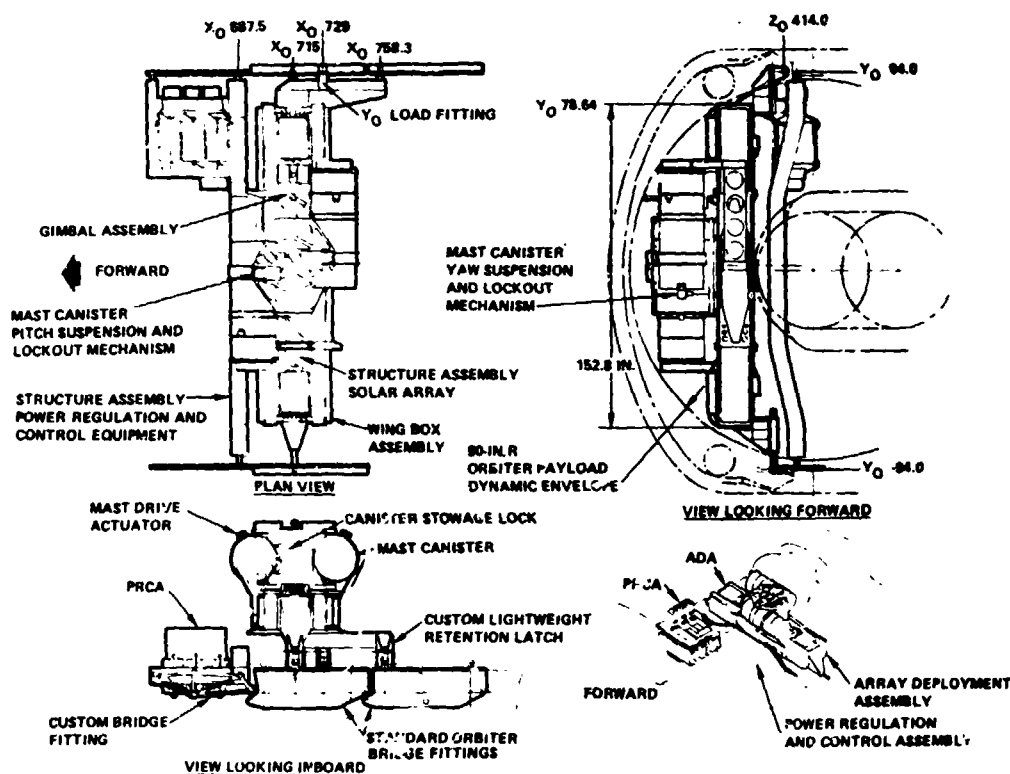
Figure 3.2.2.4-2. PEP Reference Configuration Installation

3.2.2.4.1 ADA Structure

The referenced design of the ADA structure assembly is ill crated by Figure
3.2.2.4-3. The truss structure is machined from 2219-T87 aluminum plate stock.
The beam is assembled using lock bolts and is 17 in. deep, 12 in. wide, and
140 in. long.

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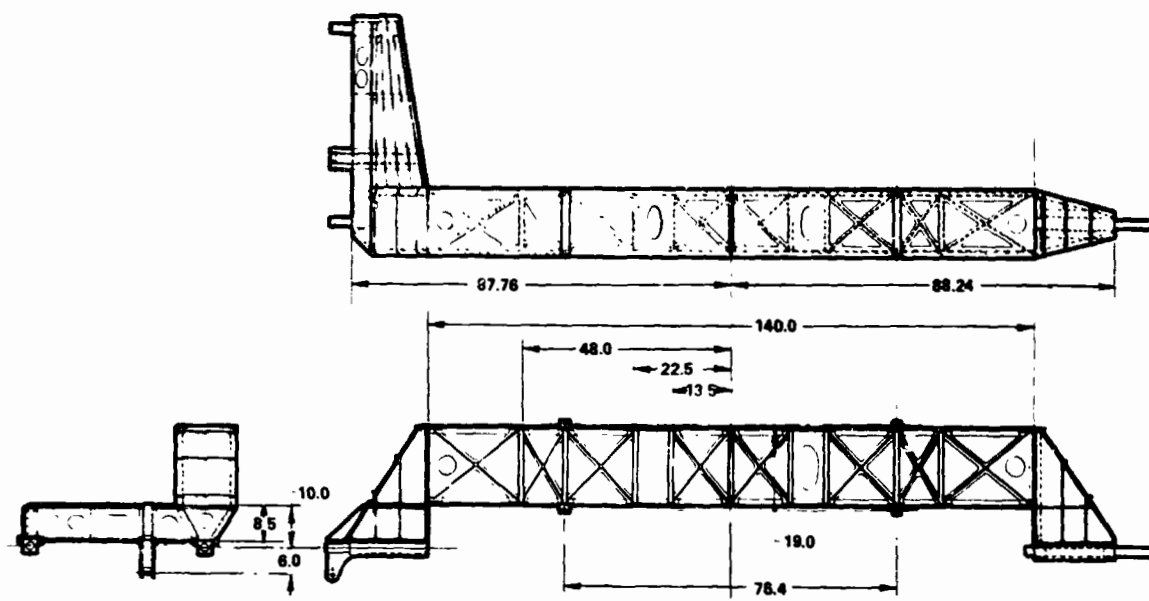
Figure 3.2.2.4-3. Structure Assembly Solar Array Support

The two wing boxes containing the array blankets are attached to the sides of
the ADA structure assembly, using a four-bolt attach pattern. The mast
canister suspension and lockout mechanism is bolted to the top surface of the
beam together with the gimbal assembly. The mast canister suspension st · ;-
ture, like the ADA beam, is a truss-type design using machine 2219-T87 alumi-
num plate stock. Attached to the suspension structure are the trunnion bear-
ings for both masts, suspension springs, lockout mechanism, and stowage
latches.

During Orbiter transport, the mast canisters are stowed with their centerline
parallel to the array wing boxes. Rotation of the canisters for deployment is
caused by initial mast extension motion through pivot and linkage arrangement.
Continued extension of the mast operates the cover latches on the wing boxes
and deploys the array blanket. The procedure is reversed for retraction and
stowage.

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3.2.2.4.2 PRCA Structure

The referenced design for the .RCA structure is illustrated by Figure
3.2.2.4-4. The structure is a square tubular design using welded sheet
2219-T87 aluminum. During all of the initial phases oı design information
available indicated that the igloo, for the all-pallet Spacelab, was below
Zo=400. Because of this, the PRCA bay spanning beam was designed and analyzed
as a straight member with the bottom surfaces at Zo 403.5. "' hₐₗ ˙ᵉcently
been learned that the dynamic envelope for the igloo asseмol. extendₛ up to Zc
412.5. As a result of this finding, the PRCA beam was modified to include an
arched center section which will clear the igloo envelope. Figure ϳ.2.2.4-5
illustrates the PRCA component arrangement and cable installation. Det ıl
design may include a cover over the top of the voltage regulators еɪ.. cold
plates to provide thermal and damage protection. The power-distribution box
would remain exposed to allow passive thermal control.
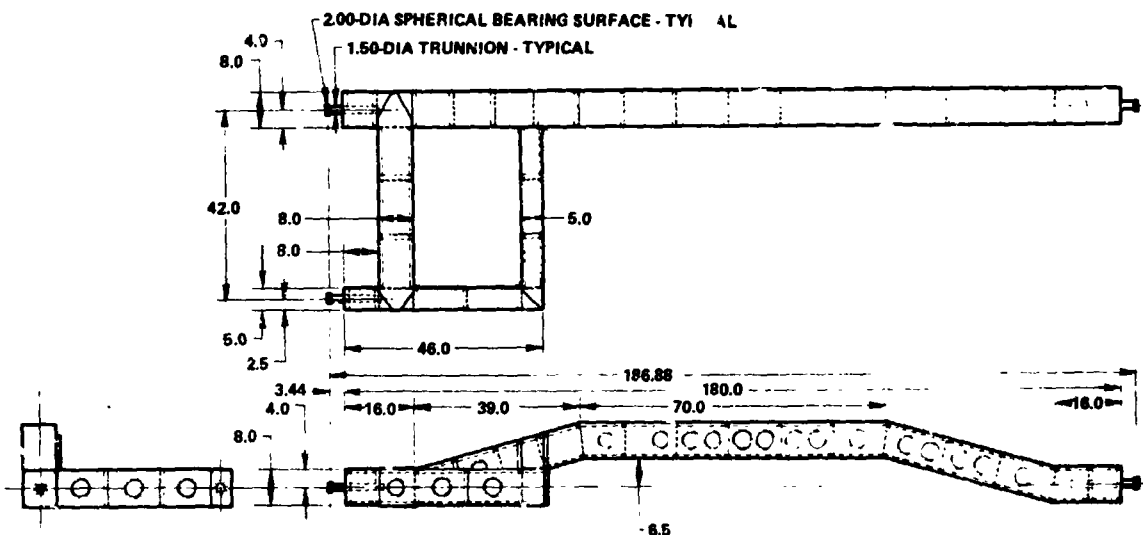
F⁓) 3.2.2.4-4. Structure Assembly Power Regulation and Control Equipment

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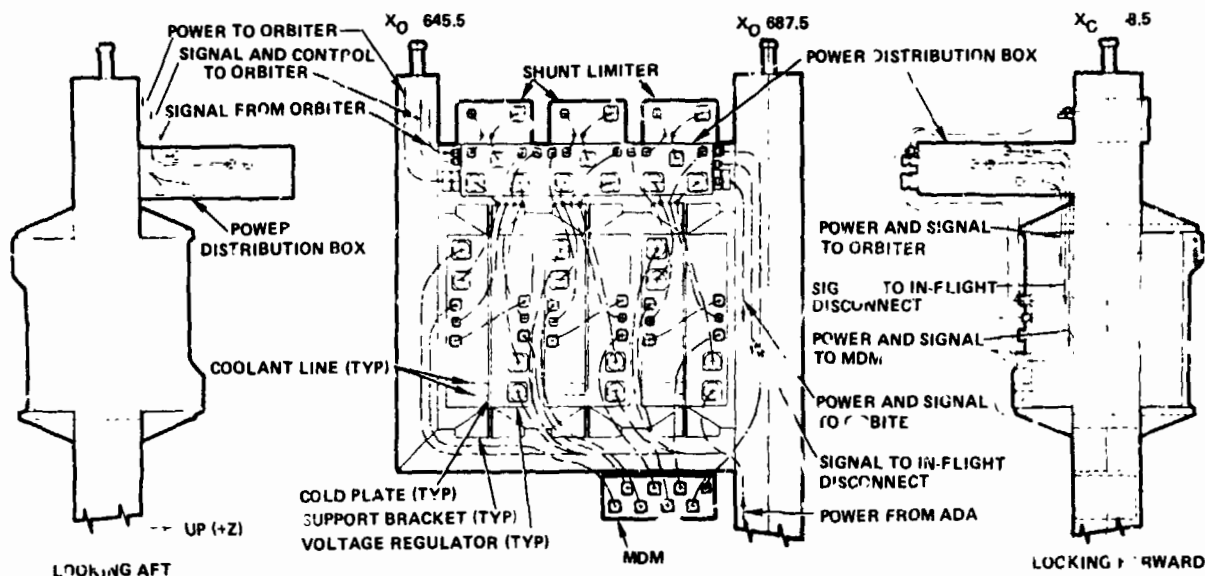
Figure 3.2.2.4-5. PRCA Component Arrangement

3.2.2.4.3 Retention Provisions

The PRCA is mounted in the Orbiter using three trunnions and two custom-designed bridge fittings. The ADA is mounted using three trunnions and three custom-design lightweight retention latches. The latches mount on either standard or custom bridge fittings which are shared with Spacelab and the short-access tunnel. For both the ADA and the PRCA, the lateral (Yo) loads are reacted by the Orbiter longeron bridge fittings.

3.2.2.4. Mechanisms

The referenced PEP design contains the following mechanisms:

    mast assembly.

    ctuator, mast drive.

    ast canister suspension and lockout mechanism.

● Gimbal assembly.

● Payload retention latch.

● Wing box cover latch assembly.

● Blanket tension mechanisms.

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The mast assembly is a continuous longeron open-lattice structure that deploys and retracts into a compact cylinder by elastically coiling the longerons. The mast assembly is illustrated by Figure 3.2.2.4-6. Battens and diagonals form evenly space bays along the deployed length of the three longerons. At the intersection between the batten and the longeron, roller lugs are attached. These lugs engage a nut in the canister which drives the mast in and out of the canister.
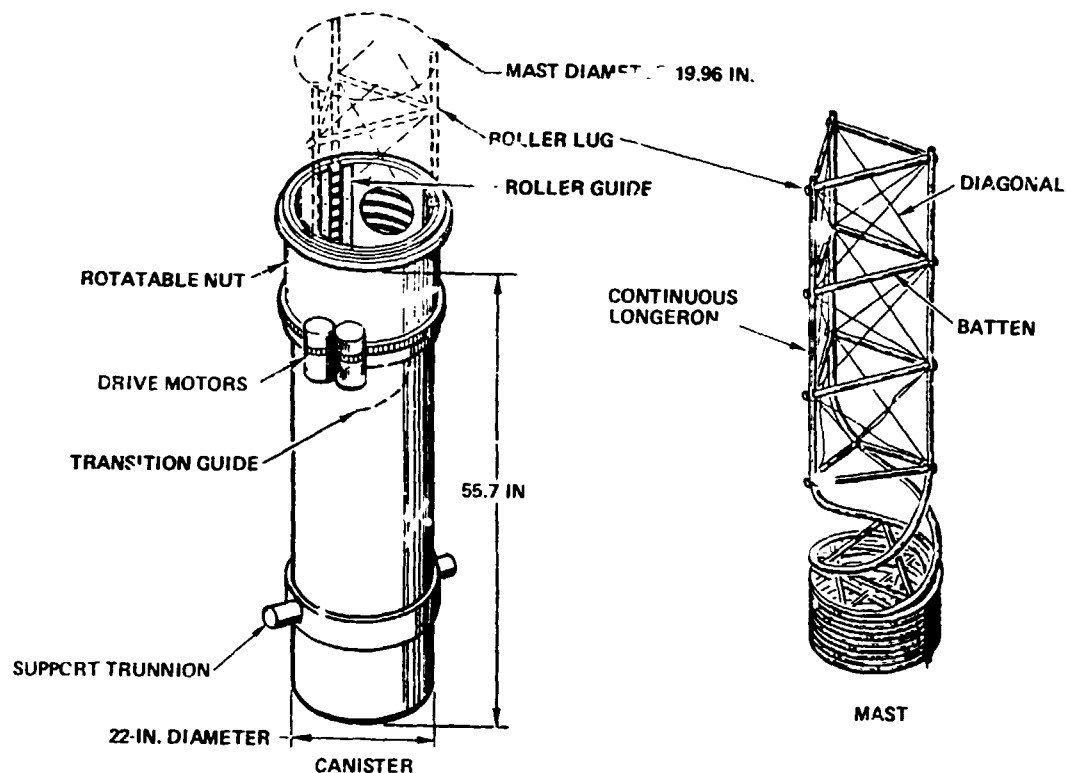
Figure 3.2.2.4-6. Mast/Canister Assembly

The mast drive power is supplied by a redundant motor, two-speed actuator. The actuator, which has a manual override, performs the basic function of deploying and retracting the mast; the initial mast motion during extension is used to rotate the canister and unlock the wing box cover. The combination of relatively fast deployment and retraction of the mast, the high locking loads of the wing box cover, and the desired slow rotation speed of the canister requires a two-speed mechanism. Figure 3.2.2.4-7 illustrates an actuator con-
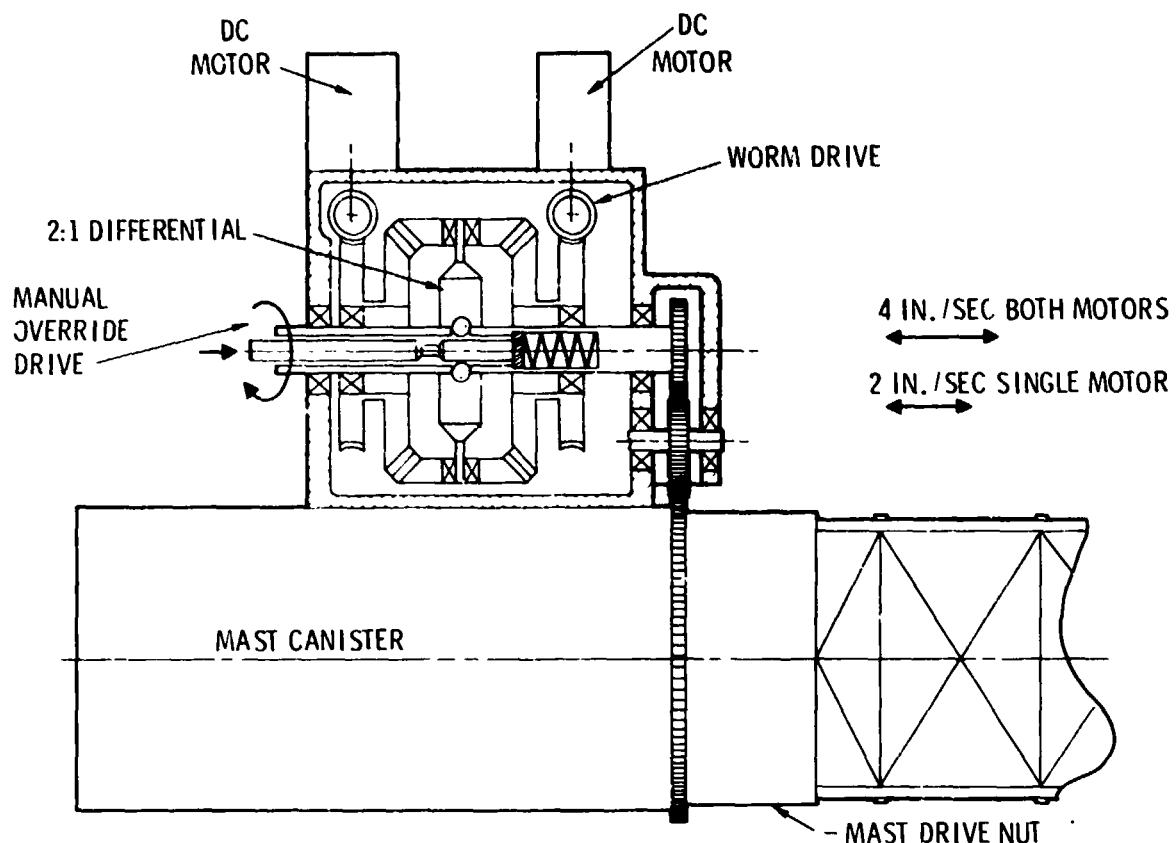
Figure 3.2.2.4-7. Mast Drive Concept — Schematic

cept using dual motors driving into a differential. By operating both motors
the actuator output runs at high speed. By operating only one motor the output
speed is reduced to one-half but the torque output is the same. It if is nec-
essary to have a speed change of greater than 2 to 1, a solenoid gear changer
could be added between the differential and the output shaft. Manual override
is provided by uncoupling the output shaft from the differential and cranking
the shaft directly.

Figure 3.2.2.4-8 illustrates the mast canister suspension concept. The mast
canister is supported on two trunnions. The bottom trunnion is supported in a
fixed spherical bearing. The top trunnion is supported in a bearing attached

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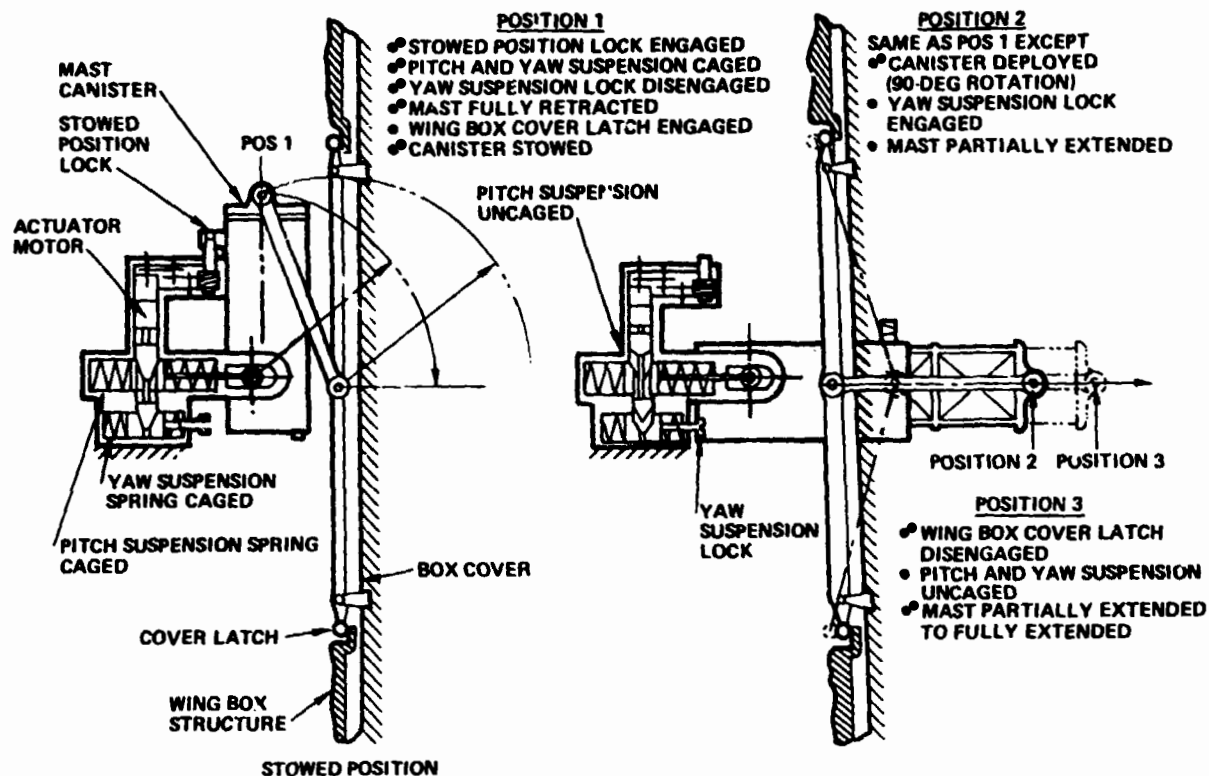
Figure 3.2.2.4-8. Schematic of Mast Canister Suspension and Self-Rotation Concept

to a spring bungee and guided in a slot. This arrangement allows the canister and mast to rotate against the spring in a plane perpendicular to the plane of the deployed array. When the canister is in the deployed position, a solenoid-operated latch locks it to a second bungee which allows rotation against the spring about the axis of the canister trunnions and in a plane parallel to the plane of the deployed array. Both springs are caged when the ADA is stowed in the Orbiter bay, during rotation of the canister, and during wing box latch operation. Electromechanical actuators are used to cage the springs.

The two-axis gimbal assembly is bolted directly to the top surface of the ADA support structure and provides continuous rotation of the deployed array in the Alpha axis and 90-deg rotation in the Beta axis. Additional description and concepts are contained in Section 2.9 of Volume 3.

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The custom lightweight retention latches are designed to interface with standard Orbiter bridge fittings and to use the standard electrical connection. Additional description and concepts are contained in Section 2.3 of Volume 3.

The wing box cover latches are actuated by motion of the mast through the interface linkage. The latches hold the folded array blanket in a compressed state at approximately 0.5 psi pressure to stabilize and protect the array blanket during transport to and from orbit.

The blanket tension for the deployed array is provided by negator spring motors, which apply torque to a cable take-up reel. The cable from the reel is attached to the array blanket through a linkage and a series of load-distribution springs. A similar mechanism is used for the two guide wires.

3.2.2.4.5 Alternate Configuration

Figure 3.2.2.9 illustrates a PEP installation using an alternate design for the ADA proposed by Lockheed Missiles and Space Company (LMSC). The general
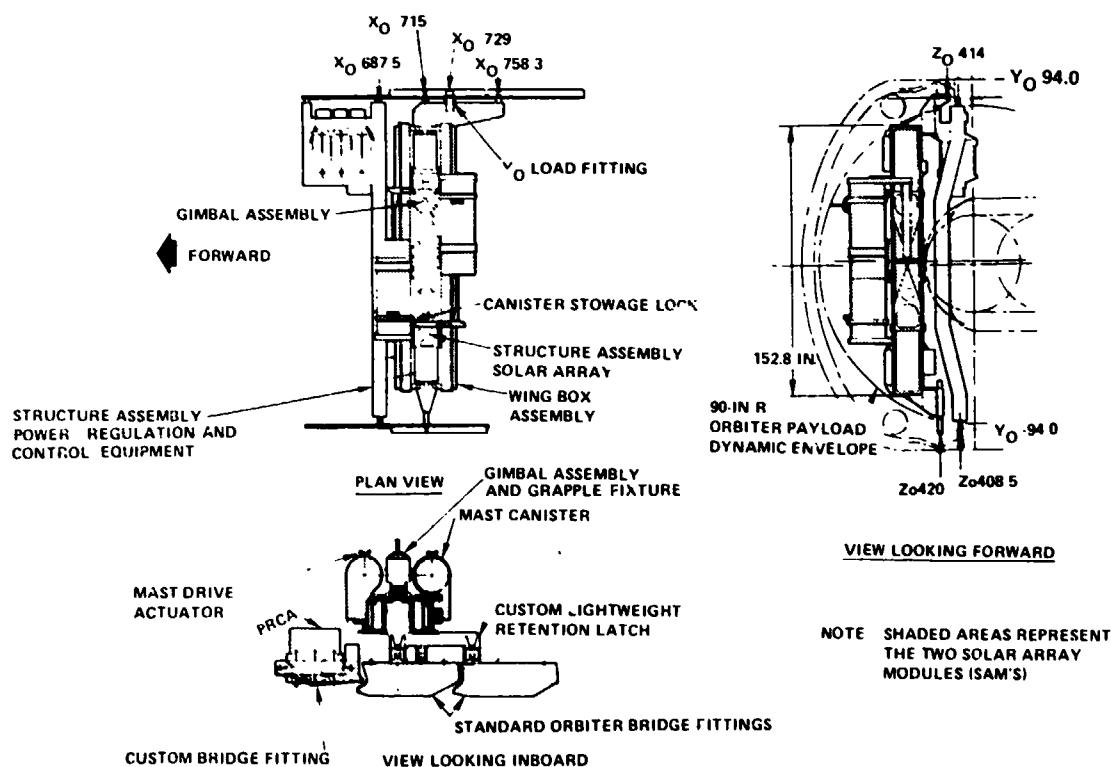
Figure 3.2.2.4-9. PEP Installation Using LMSC Solar Array
Module Concept

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arrangement of this configuration is the same as the previously described reference design. The difference lies in the mechanization of the mast canister support on the ADA. This ADA design assembles the wing box and the mast/canister together as a unit and refers to the assembly as a solar array module (SAM). The concept is the same as the modular wing design (Configuration 4), and discussed in Section 3.2.2.3.

The SAM interfaces with the array support structure assembly using a four-bolt pattern. The mast canister is structurally supported by a single cantilever trunnion attached to the wing box. A stowage lock is attached to the wing box and engages and locks the canister in the stowed position for transport. The mast drive actuator has two motors. One motor drives the mast at the high extension/retraction rates, and the second motor is used for low speed and high force levels required for latching the wing box. Each motor has a dual winding to provide electrical redundancy. The mechinization of the spring compliance for array suspension has not been defined, but several design options may be considered. These options are:

A. The compliance springs installed in the canister trunnion support between the canister and wing box.

B. The compliance springs integrated into the mast, and canister design and provide relative motion between the mast and the canister.

C. To hinge the SAM on the support structure and provide a spring between the SAM and array support structure as illustrated by Configuration 4 in Section 3.2.2.3. This arrangement provides only one axis of freedom; the other axis of compliance would be incorporated in the canister trunnion pivot.

Option B has been proposed as a concept by LMSC and AEC Able Engineering. Details of the mechanization of this concept have not been revealed; however, if it were shown that the mechanization would not unduly complicate the mast design it would represent an attractive alternate.

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3.2.3 Avionics and Control Subsystem

The PEP avionics and control subsystem performs test, configuration, deploy/stow, and array pointing functions. When installed in the Orbiter payload bay, it is supported by the Orbiter's systems management general-purpose computer (GPC), the multifunction CRT display system (MCDS), the RMS, and the remote payload latch release system. The use of capabilities afforded by these existing facilities permits limiting PEP-peculiar electronic assemblies to: (1) a multiplexer/demultiplexer assembly (MDA) in the bay for acquiring data canister rotation and mast extension/retraction and gimbal control, (3) a sun sensor with processor for providing sun pointing error signals, (4) a bus coupler for interfacing with the GPC data bus.

3.2.3.1 Avionics and Control Subsystem Requirements

The functional requirements for PEP are listed in Table 3.2.3-1. They consist of operations which the equipment must perform during array deployment and

Table 3.2.3-1. Avionics Functional Requirements

- Activate/deactivate PEP pointing and control electronics equipment.

- Engage/disengage the power-on connector(s) between the RMS end effector and PEP grapple fixture.

- Do mast-canister rotation and mast extension/retraction.

- Acquire and track the sun.

- Maintain the array axes in selected angular positions and/or rates in respect to the sun line and/or end effector axes.

- Select regulator and power-distribution box configurations, i.e., connection/removal of power-conditioning equipment from power buses.

- Set regulator/limiter voltage levels.

- Disconnect the Orbiter/PEP power bus interface to allow RMS jettison.

- Display PEP command status.

- Monitor and display PEP equipment operation and status parameters.

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while maintaining the array face (solar cells) perpendicular to the sun or at commanded angular positions in respect to the Orbiter or its velocity vector.

Table 3.2.3-2 identifies basic performance requirements; they are primarily concerned with gimbal control. Since array power output is proportional to the cosine of the angle between a line normal to the array and the sun line, performance loss is small for rather large angular errors. As a result, performance requirements are not severe.

Table 3.2.3-3 identifies software requirements. They are split between those imposed on the system management computer software and those processing data in PEP equipment. It is seen that systems management requirements are not extensive, being primarily devoted to control and monitoring of the PEP via the MCDS equipment. The PEP software performs array control algorithms and operation sequencing.

The avionics interfaces with the RMS and CPC result in interface requirements being imposed upon PEP equipment. Power consumption of ADA electronics has

Table 3.2.3-2. Avionics Performance Requirements

- Track the sun within 2 deg.

- Slew to defined Alpha and Beta gimbal positions (usually used for initialization) at rates from 0 to + 0.5 deg/sec.

- Slew at defined angular rates for an indefinite time.

- Command gimbal postions and slew rates via the MCDS keyboard, ground links, or preprogrammed table.

- Stop slewing at any time by keyboard or ground link command.

- Operate with sun-sensor feedback in either gimbal axis or both.

- Maintain a gimbal position error no greater than 12 deg during loss of a "sun-presence" signal.

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Table 3.2.3-3. Systems Management and PEP Software Requirements

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<u>Systems management software functional requirements</u>

● Acquire, process, and display PEP data.

● Format and output PEP commands.

● Control regulator connection to main power-distribution assemblies.

● Set regulator sense voltage levels.

<u>PEP software functional requirements</u>

● Effect data transfers.

● Sequence ADA mast extension and retraction.

● Perform manual and automatic PEP mode control.

● Acquire array deployment assembly data.

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been limited to about 224 watts at 24 VDC and up to 314 W at 32 volts input at the RMS shoulder due to wire resistance of the SPEE cable harness. In addition, SPEE wires have been allocated to array deployment assembly functions such as power connector actuator control and the CEA data bus. The requirements for data interchange with the GPC resulted in the addition of a data bus coupler to establish a compatible interface with the Orbiter's bus terminations. A spare AFD bulkhead connector connects PEP power regulation control equipment to a continuation of the SPEE cable harness and provides access to an on/off switch for control power.

Existing Orbiter provisions for latch control of the ADA latch trunnions are used. They include the payload select switch and toggle on panel A6, the latch discrete status shown on the RMS control display format, and the actuation circuitry ending in dual connectors per latch providing AC and DC latch power.

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3.2.3.2 Avionics and Control Subsystem Performance

Upon crew request, the MCDS and GPC acquire data issue commands via the MDA to power regulation and control equipment located within the payload. The standard bus rate of 1 MBPS is used. A lower rate (30 to 50 kbps) is used between the MDA and CEA to eliminate the need for controlled bus wire and allow the use of SPEE wiring without modification. Off-the-shelf MDA modules have been used to provide 48 28-VDC discrete outputs to control in-bay equipment, 32 differential inputs for data acquisition, and 8 serial input/output channels for regulator control (only 6 are presently used). These provisions allow regulator configuration (connection of regulators to main power distribution assemblies) with a single command input from the MCDS keyboard. One command per regulator is required for level setting although software may be structured to set all levels with a single command provided they are known and entered prior to commanding a value change. Within the CEA, similar provisions exist for data acquisition and additionally, for conditioning signals reflecting the status of the arrays (10 line pairs/wing) and other ADA instrumentation. Measurements are sampled once per second, a rate selected because of control rather than data-acquisition requirements.

Four modes of PEP control are available, two manual and two automatic. In the manual mode, array position commands may be entered to slew the PEP gimbals to a given position; rotation is limited by ADA interference to 0 to 90 deg for the Beta gimbal and is (continuous) 0 to 360 deg for the Alpha gimbal. This mode also permits establishing constant gimbal rates: a maximum rate capability of $\pm$ 0.5 deg/sec, or double the 0.25 deg/sec Orbiter step rate permitted with acceptable mast loads. In the automatic mode, sun track submode, both gimbals are controlled using sun-sensor error signals. The track submode allows "feathering" the array in one axis while controlling the other axis with the sun sensor. The pointing accuracy requirement of $\pm$2 deg permits limiting the projected area of the array face in the direction of travel to 3% and reduces worst-case power loss to less than 0.1% in sun track.

An on-off controller was selected in preference to a proportional controller due to its simplicity and because the proportional controller did not provide damping in all requisite axes. Structural damping is provided mechanically. Figure 3.2.3-1 is a functional block diagram of the selected concept. The portion
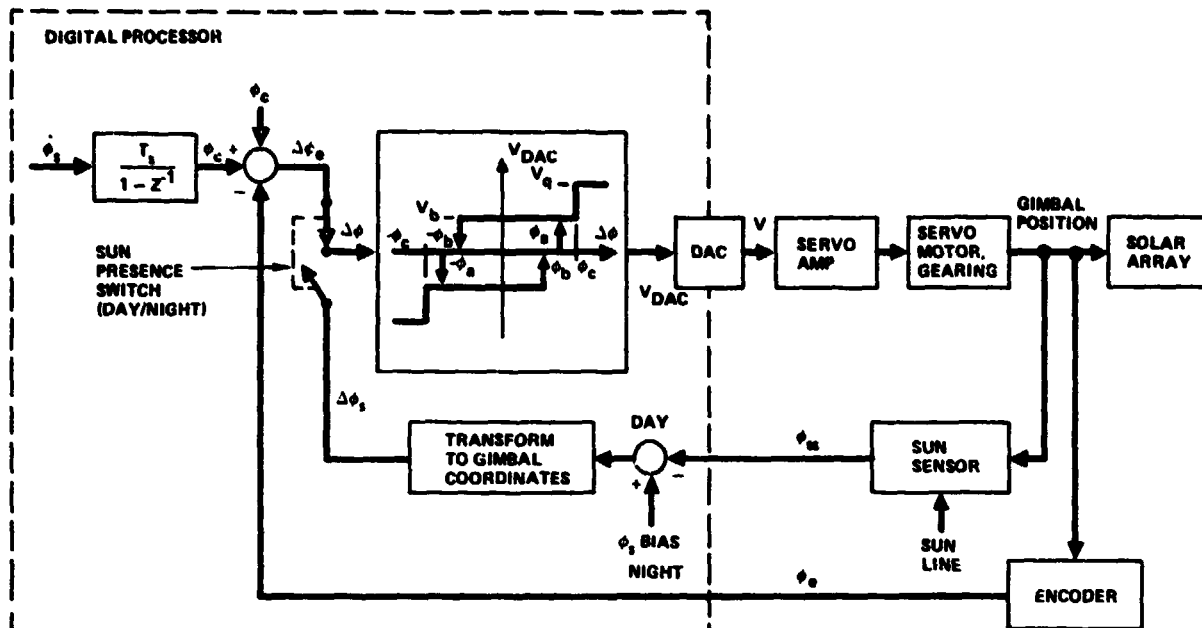
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Figure 3.2.3-1. Block Diagram of On-Off Controller

of the diagram shown in dashed lines would be implemented in the CEA micro-processor. Commanded rates ($\dot{\phi}$s) or positions (ϕc) are input to the servo drive voltage generation block (VDAC-D/A converter voltage) as position errors after subtraction of encoder-generated shaft position. Sun sensor error signals would be used directly after coordinate system transformation in the automatic sun-track mode.

An execution rate for the control algorithm of once per second has been judged acceptable since the array is driven through high-compliance springs with a response time of 0.02 Hz. The achieved gimbal rate can be set near the required rate, and the array smoothly tracks the reference with occasional high or zero achieved rate periods occurring to adjust the average achieved rate to the required rate. The constant rate is desirable since structural resonance excitation is minimized. The controller is effectively an open-loop controller for long periods of time, which can eliminate stability problems associated with structural resonances.

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3.2.3.3 Major Trades and Analyses

Trades and analyses performed in support of PEP definition were varied, running the gamut from evaluating the configuration to analyzing the electromagnetic environment. The following paragraphs indicate the subject of these tasks and present their conclusions. More definitive task treatment may be found in Volume 3.

A. Task 2.14, Solar Array Control Avionics Requirements/Criteria Definition. Established that a proportional or on-off array controller (diagram shown in Figure 3.2.3-1) can be used to satisfy pointing requirements. The latter, due its simplicity, is recommended as the baseline.

B. Task 2.15, Controls System Management, GPC and Array Control Processor Interface Definition. Defined the software modules required in the SM computer and concluded that they were few in number. It also defined the control and status displays to be presented on the MDCS and requirements for the CEA microprocessor software.

C. Task 2.18, Orbiter DAP Utilization Evaluation. Limit cycles resulting from use of primary thrusters impose excessive loads on PEP with the exception of the array located ahead of the Orbiter's nose. Firing of the vernier system does not result in loading problems during nominal limit cycle operations. Limited Orbiter maneuvers are possible with either the PRCS or VRCS.

D. Task 2.19, Pointing/Control Avionics Concept and Operations Analyses. Developed microprocessor implementation concepts and software flows.

E. Task 2.20, EMC Analysis. Established the radiated field intensity levels impinging on PEP due to Orbiter transmitters and antennas.

F. Task 2.21, Alternate Solutions for PEP/RMS Control and Drive Power Wiring. Traded the relative merits of switching existing SPEE wiring vs the addition of PEP-peculiar wiring harnesses and recommended retention of the former concept.

3.2.3.4 Avionics and Control Subsystem Description

The PEP avionics subsystem is illustrated in Figure 3.2.3-2. That part of the PEP avionics subsystem located on the PRCA consists of an MDA connected to a data bus coupler that interfaces with the GPC data bus terminations in the payload bay. The deployed position of the avionics subsystem on the ADA

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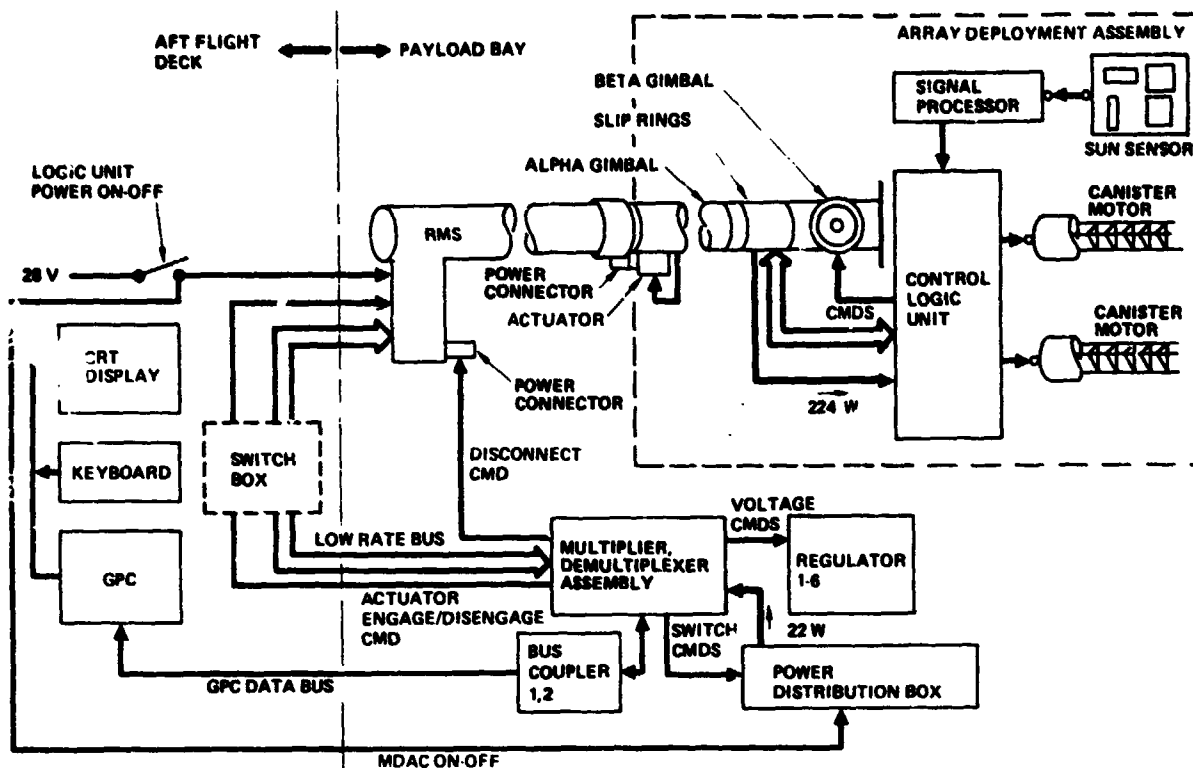
Figure 3.2.3-2. PEP Avionics Block Diagram

interfaces with RMS SPEE wiring. It consists of the CEA, a sun sensor, and
sun signal processor. The CEA controls array canister rotation and mast
extension/retraction, provides gimbal drive signals and acquires/conditions/
formats ADA instrumentation signals. A switch box in the Orbiter (not part
of PEP) allows sharing of the RMS wiring with payloads. The use of existing
Orbiter/RMS wiring for control purposes is more clearly depicted in Figure
3.2.3-3. All wiring to the on-orbit station is currently installed. The
toggle switch, located on the standard switch panel, allows power on-off
control to the PEP avionics equipment. The remaining cables between the on-
orbit station and the power interface would be new.

The MDA includes a multiplex interface adapter, sequencer control module,
and sequencer memory for interfacing with the GPC bus. Other modules consist
of differential DC input, 28-VDC single-ended discrete input and output
modules, serial I/O modules, and a special low-rate bus module. All but the
latter would acquire data from, and input commands to, the power regulators
and power-distribution box. The low-rate bus module would be used for
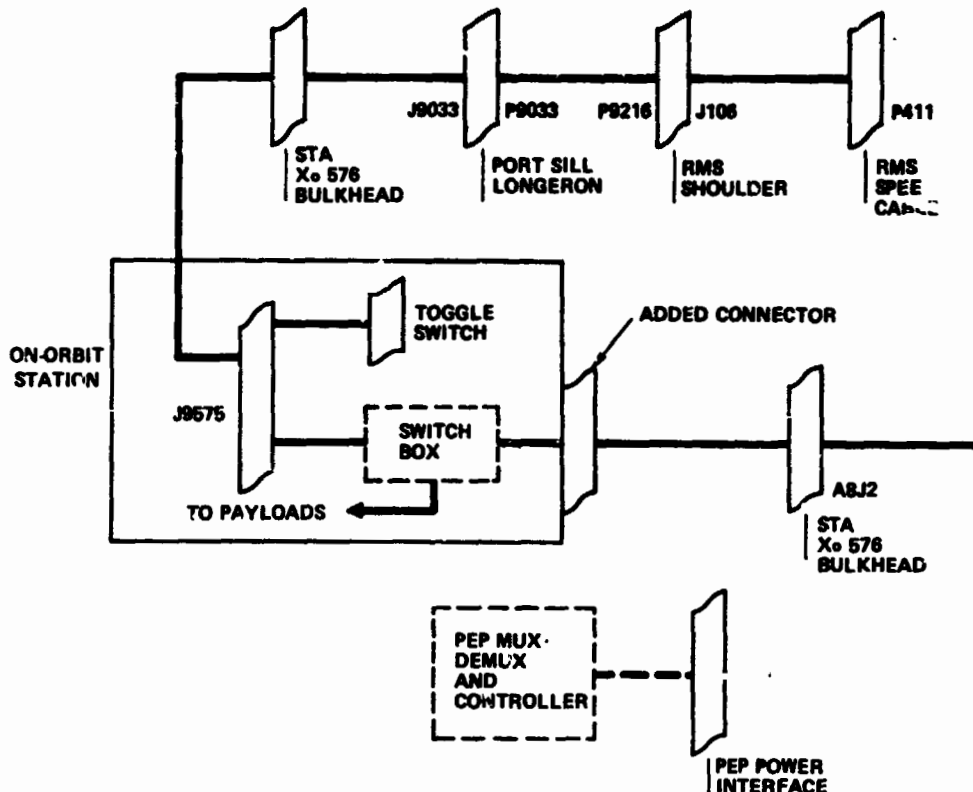interfacing the CEA.

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Figure 3.2.3-3. RMS Wiring/PEP Interface

The CEA contains dual microprocessors (one on standby) for signal processing
ar⁻ ₌bal control. Instructions stored in memory or input through the MDCS
ard are transformed into commanded rates by the pointing software
₌richms and switching logic. Error signals provided by the sun sensor
₌ring sun presence or by shaft encoders provide position feedback for array
rate and position control.

All control inputs are presently provided by the MCDS keyboard with the
exception of power on/off. Failure analyses studies, to be conducted in the
future, may also require hardwired redundant controls for operating the power
connector actuator release on the grapple fixture and for breaking the power
harness at the RMS shoulder should the RMS require jettison. Display of control
inputs via the MCDS CRT is illustrated in Figure 3.2.3-4. Formatting of this
display, as well as the processing and display of PEP status, is performed by
the systems management computer.

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┌─────────────────────────────────────────────────────────────────┐
│      XXXX/XXX          PEP CONTROL                                │
│                                                                   │
│    PEP PWR      XXX          PWR CONN XXXXXXX    CAN ROT  1XXX  2XXX │
│                                                                   │
│    AUTO/MANUAL       XX                          MAST EXT 1XXX  2XXX │
│                                                                   │
│              GIMBAL POSITION                  INFLIGHT DISCONNECT  │
│                                                    1      2        │
│    ALPHA     XXX                            ARM    X      X        │
│                                                                   │
│    BETA      XXX                            FIRE   X      X        │
│                                                                   │
│              GIMBAL RATE                                          │
│                                                                   │
│    ALPHA       XX                                                │
│                                                                   │
│    BETA        XX                                                │
│                                                                   │
│              REGULATOR ADJUST LEVEL                              │
│                                                                   │
│      1 XX.XX    2 XX.XX    3 XX.XX    4 XX.XX   5 XX.XX  6 XX.XX   │
│                                                                   │
│              PDB SWITCH SETTING                                  │
│                                                                   │
│        1 X 2 X 3 X 4 X 5 X 6 X                                   │
└─────────────────────────────────────────────────────────────────┘
```

Figure 3.2.3-4. PEP Control Display Format

3.2.4 Thermal Control Subsystem

The PEP thermal control subsystem, in conjunction with Orbiter capability,
provides cooling of all heat generated within the Orbiter, PEP, and payloads.
These heat loads include electrical power dissipation, PEP and Orbiter para-
sitic loss, Orbiter fuel cell waste heat, and metabolic loads.

The reference thermal control subsystem design is a result of several trades
which addressed key issues in the study. The issues related to adequacy of
Orbiter heat rejection and collection when operating in PEP mode and to PEP
thermal control configuration, i.e., active versus passive cooling of PEP vol-
tage regulators.

A brief description of trades and analyses performed in the study are pres-
ented in the following paragraphs. Requirements, performance, and a descrip-
tion of the subsystem are also included.

3.2.4.1 Thermal Control Subsystem Requirements

Table 3.2.4-1 lists key design requirements for the PEP thermal control sub-
system. Detailed heat loads delineated from information in Table 3.2.4-1 are

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Table 3.2.4-1. Thermal Control Subsystem Requirements

Mission

Orientation	All, especially earth viewing, stellar, solar and low gravity
Inclination	28.5 to 104 deg
Crew size	4
Altitude	100 to 600 nm

Performance

Heat rejection (in conjunction with Orbiter)	Waste heat at 21 to 29-kW power level
Orbiter power level	14 kW
Orbiter power rejected by radiator	80% of 14 kW
Flash evaporator system performance limit	30 lb/hr (13.6 kg/hr)
Water storage limit	250 lb (113.6 kg)

Heat Collection

Maximum fuel cell return temperature	140°F (60°C)
Maximum midbody cold plate temperature	120°F (48.9°C)
Maximum regulator temperature	150°F (65°C)
Maximum avionics box Temperatures	104 to 176°F (40 to 80°C)

Reliability and safety

Redundancy	Maintain separation between primary and secondary loops

EVA compatibility

EVA routes	Minimize intrusion
Loads	Tolerate EVA crew kickoff loads

Orbiter fluid loop compatibility

Pressure drop	2.6 psig (1,831 kg/m^2)
Fluid	Freon 21
Inlet temperature	74°F (23.3°C)
Pressure	Same as Orbiter
Cleanliness	Same as Orbiter

given in Figure 3.2.4-1 for two levels of power, 21 and 29 kW. All loads shown are instantaneous loads; no thermal inertia or transport lag is assumed in the system. Shade-side heat loads are the same for PEP and fuel-cell-powered Orbiter, but the PEP loads are 10.5 kW less at 29-kW power load for sun operation when heat from fuel cell waste is reduced. The result is an orbital average PEP load of 6.6 kW less at low Beta angles.

3.2.4.2 Thermal Control Subsystem Performance

The reference PEP thermal control subsystem design relies upon the Orbiter for heat rejection and as such the Orbiter performance is of primary interest. Performance considerations include heat-rejection capability at key orientations and water-balance influence on operation of the flash evaporator system.

Orbiter heat-rejection capabilities were determined in terms of maximum electrical power as limited by heat rejection. Basic Orbiter performance came from NASA thermal control data in terms of instantaneous heat rejection around the orbit. This data was for an altitude of 270 nm (500 km) and an 8-panel radiator configuration with forward panels deployed at 35 deg. Some performance
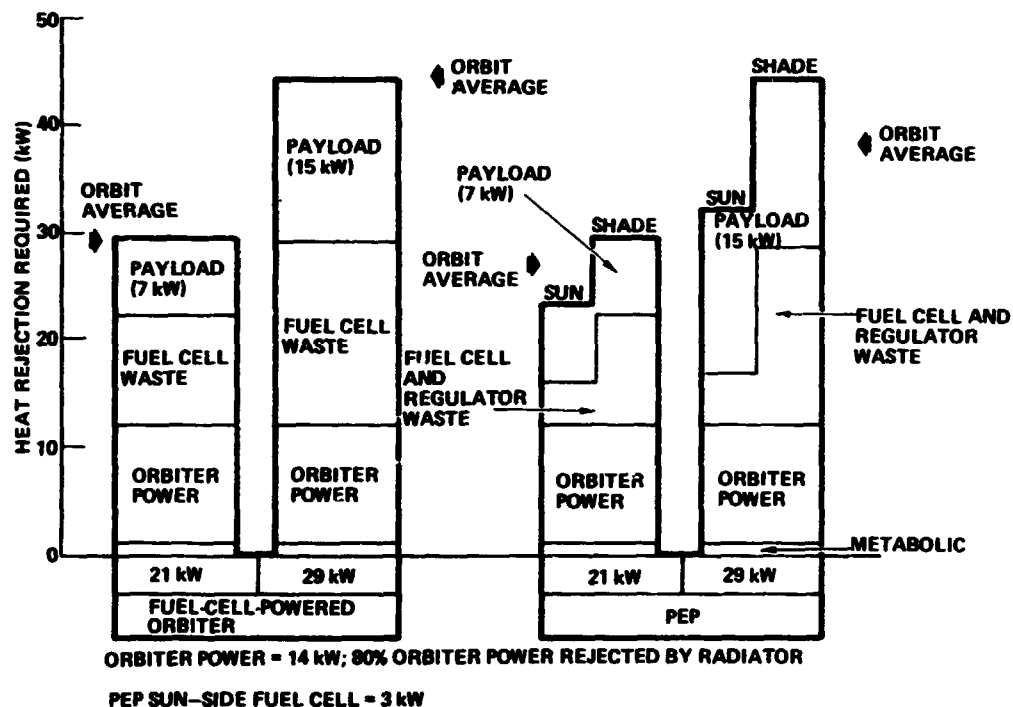
Figure 3.2.4-1. Heat-Rejection Requirement of Fuel-Cell-Powered Orbiter and PEP

improvement can be obtained for most orientations with a 60-deg radiator deployment, and heat-rejection capability was computed for this angle for a favorable orientation. Performance determination was based on average performance on sun or shade side of the orbit, depending on which side is controlling.

Figures 3.2.4-2 to 3.2.4-6 give the heat-rejection capability in terms of maximum electrical power for Orbiter and for Orbiter/PEP operating modes. Data is included for three levels of flash evaporator system operation corresponding to none, sustained, and maximum operation levels. The flash evaporator water use for sustained operation is just equal to fuel-cell-generation rate less crew use. Level of maximum performance occurs at a water-use rate of 30 lb/hr (13.6 kg/hr).

Figure 3.2.4-2 shows the worst-case performance for earth-viewing payloads, i.e., nose perpendicular to orbit plane with bay directly toward earth. This orientation results in solar influx to the radiator underside when the Orbiter is near the terminator. Data from the figure show that higher power levels of
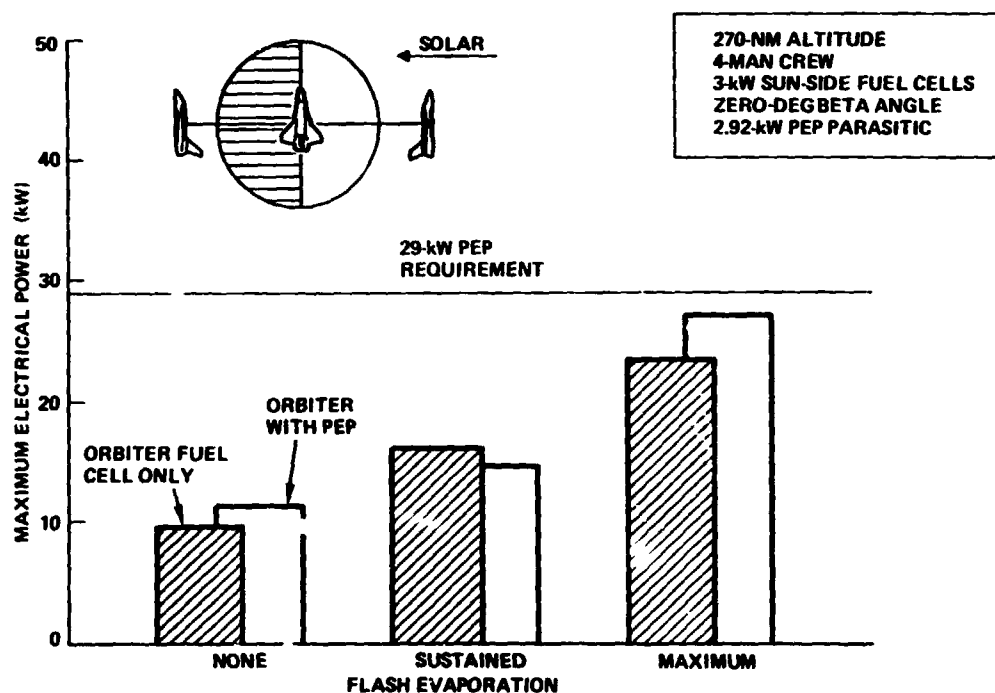
Figure 3.2.4-2. Heat-Rejection Performance, Earth Viewing

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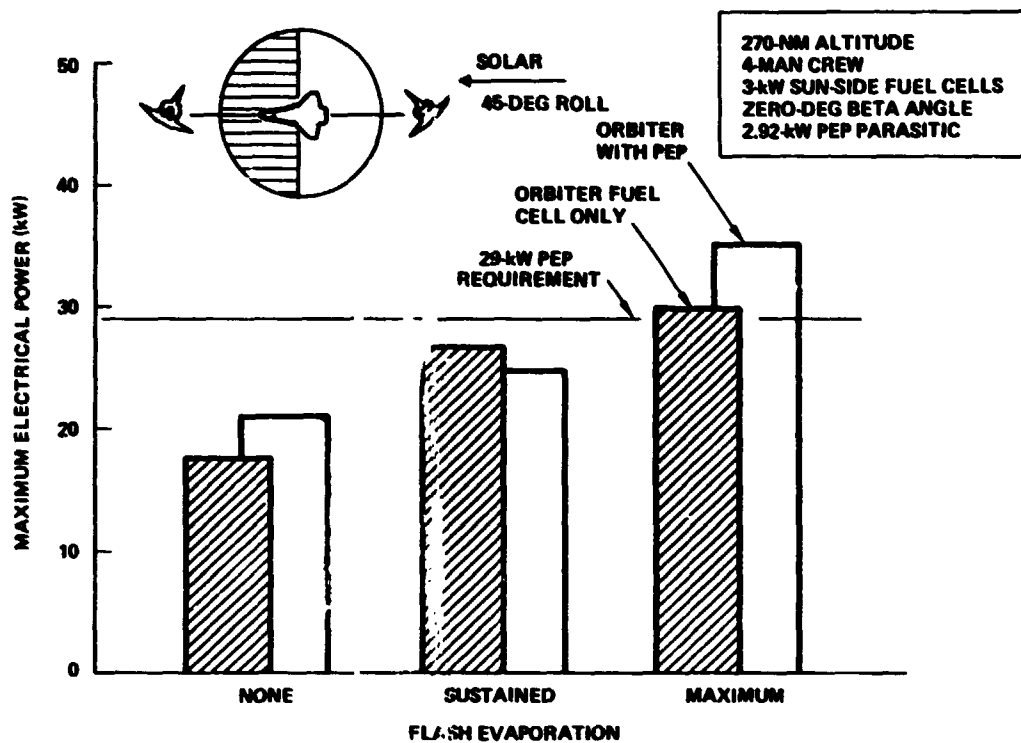
Figure 3.2.4-3. Heat-Rejection Performance, Earth Viewing (45-Deg Roll)
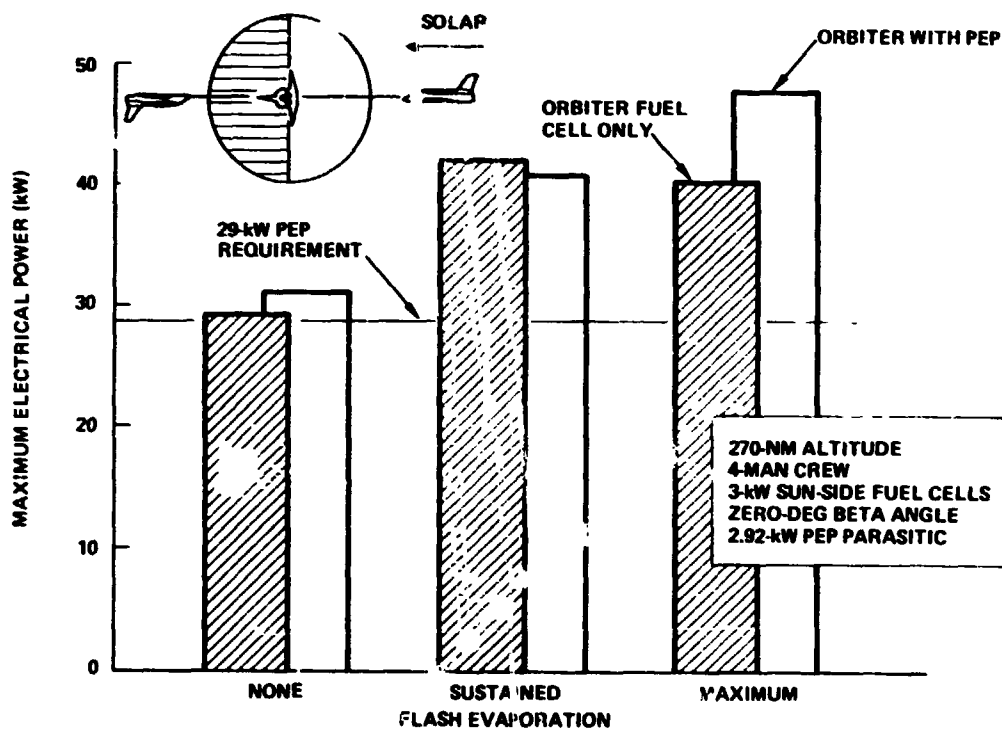
Figure 3.2.4-4. Heat-Rejection Performance, Low-g Sustained Operation

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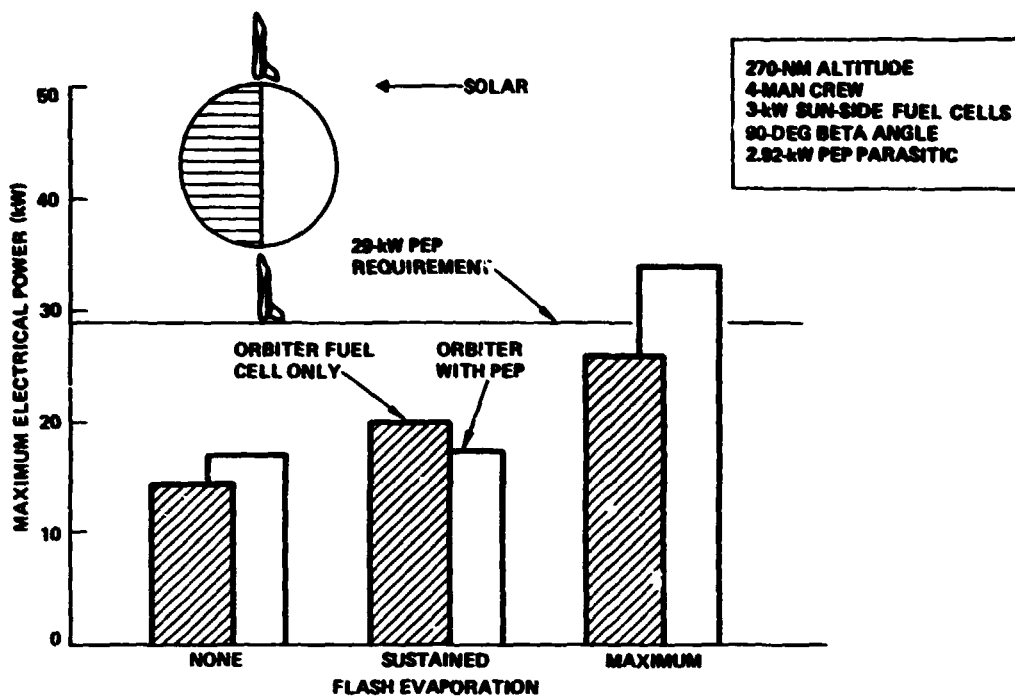
Figure 3.2.4-5. Heat-Rejection Performance, Solar Viewing
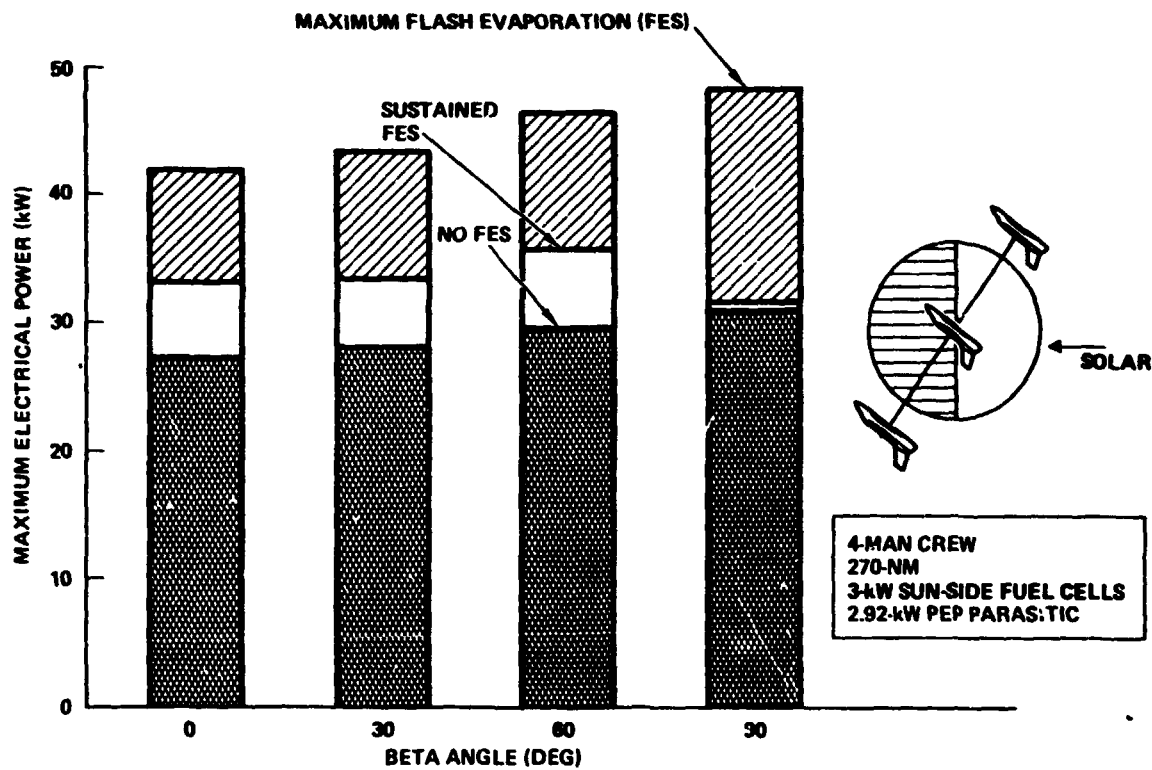
Figure 3.2.4-6. Beta-Angle Effect on Orbiter/PEP Heat Rejection Performance

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1.4 to 3.5 kW are possible with the PEP except for sustained operation, which is 1.5 kW lower. Sustained performance is lower because reduced fuel cell operation results in less water available for flash evaporator use.

Improved earth-viewing performance is obtained for the orientation shown in Figure 3.2.4-3, which is nose along velocity vector with bay 45 deg from local vertical. Higher power levels ranging from 3.3 to 5 kW are obtained with no and maximum flash evaporation, respectively. A 1.8-kW lower capability exists for sustained operation for the reason mentioned above for X-POP orientation. The Orbiter/PEP configuration can operate at significantly higher power levels than Orbiter alone. The 3.3-kW improvement for no flash evaporation is of particular significance for experiments sensitive to water vapor in the vicinity of Orbiter.

A favorable orientation is depicted in Figure 3.2.4-4, which is nose gravity gradient with constant roll to reduce solar impingement on the radiators. This type of orientation is typical of long-duration, low-gravity missions with low-RCS propellant consumption. Life science and material processing experiments fall into this payload category. Data in the figure show that a 29-kW power level can be sustained without flash evaporator system use.

Solar pointing performance at 90-deg Beta angle is shown in Figure 3.2.4-5. The power level limit is 2.7 kW higher for no flash evaporation, but the level for sustained is only 0.5 kW higher. This is because the fuel cells operate at idle during the entire orbit and only generate 1.53 lb/orbit (0.7 kg/orbit) of water for cooling use. The maximum PEP power output of 29 kW can be obtained however at higher water use rates by the flash evaporator system.

Performance for earth viewing and low-g orientations are for zero-degree Beta angle. Performance will tend to improve at higher Beta angles. This effect can be seen in Figure 3.2.4-6, which gives performance for Beta angles from 0 to 90 deg for a stellar viewing orientation. This represents bay inertially oriented with nose perpendicular to the orbit plane. The 29-kW maximum PEP power level can be accommodated at high Beta angles for no flash evaporator use. A power level of up to 27.3 kW can be accommodated at low Beta angles with no flash evaporation.

Water balance impacts available heat rejection for Orbiter due to effects on water available for flash evaporator use. Figure 3.2.4-7 shows water balance as a function of electrical power level for earth-viewing orientation with 45-deg roll. Fuel cell rate of produced water increases as power level increases. Crew needs are constant at 1.47 lb/hr (0.67 kg/hr).

Flash evaporation is required above the 20.9-kW power level as shown in the figure. The available water rate just matches the required rate at 24.6 kW. Above this power level, stored water is needed and on-board stores are being reduced. Below 24.6 kW, water is being stored. Maximum water usage occurs at 35 kW, and water rate through the flash evaporator system amounts to 30 lb/hr (13.6 kg/hr).

The Orbiter water storage is limited normally to 165 lb (75 kg). Therefore, a time limit exists for duration of operation where water is being stored or depleted.

Figure 3.2.4-8 gives time limits for storing or depleting water in the storage tank as a function of electrical power level. Data is given for both Orbiter alone and Orbiter with PEP. During Orbiter/PEP operation, water is stored
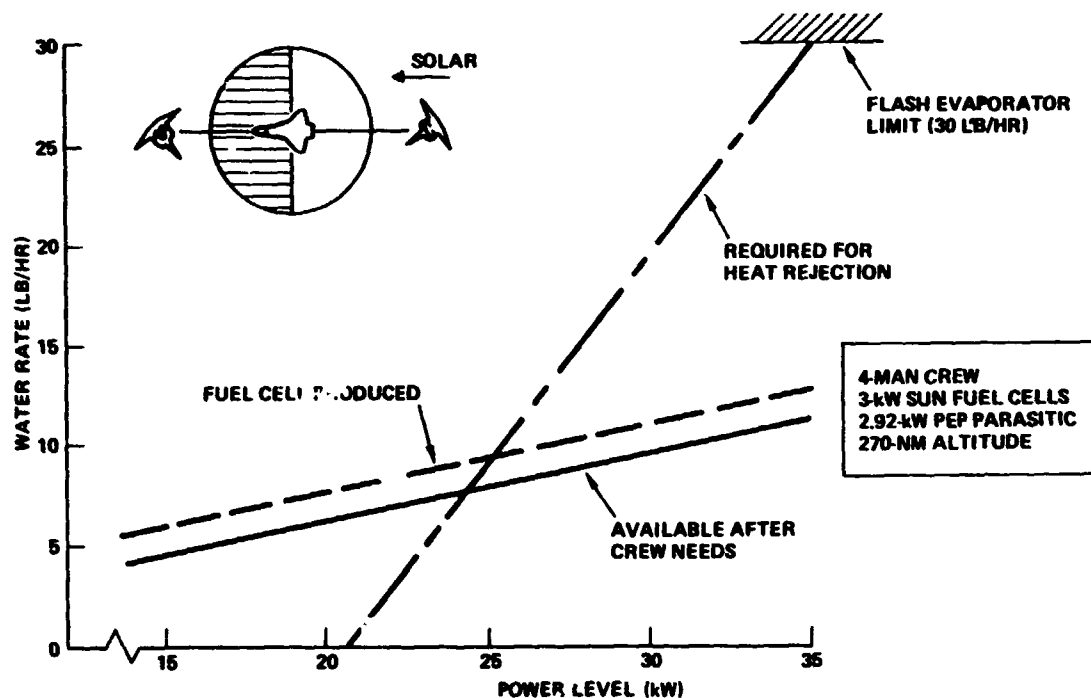
Figure 3.2.4-7. Orbiter/PEP Water Balance With Flash Evaporator Cooling

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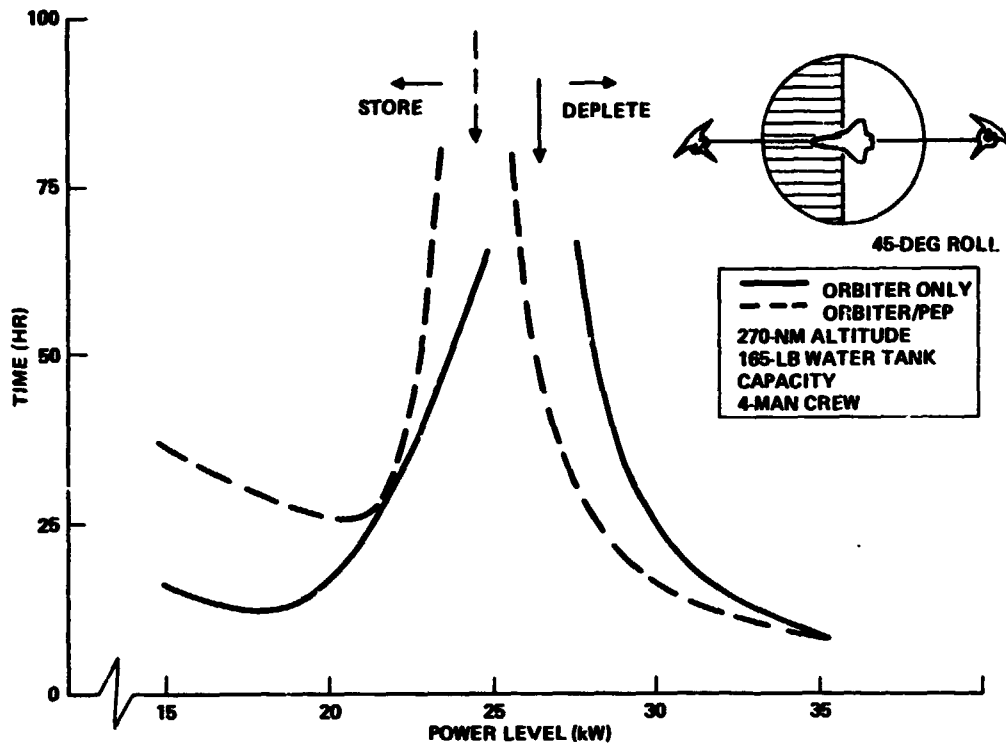
Figure 3.2.4-8. Water Store and Flash Evaporator Hold Times

below 24.6 kW where use rate is less than fuel cell production rate. As the electrical power is reduced, the water storage rate increases because the use rate decreases faster than the production rate (see slope of rate lines in Figure 3.2.4-8). Minimum storage time occurs at 20.9 kW when zero water is needed for flash evaporation. Storage time increases below this point because production of fuel cells decreases as power decreases.

Storage times for Orbiter/PEP are shorter than for Orbiter only because more water is produced relative to use requirements and because the storage time curves are shifted because of lower heat-rejection loads for Orbiter than for Orbiter/PEP.

The length of time that power levels can be sustained when water is being depleted is greater for Orbiter only. This is due to the shift in curves to the right for Orbiter-only operation because of higher heat-rejection requirements. More water is produced for Orbiter only because of higher fuel cell operating levels but more water is required because of fuel cell waste heat loads.

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Each orientation has a characteristic curve such as that shown in Figure 3.2.4-8. The difference in each curve will be primarily a shifting to right or left corresponding to the point the where net water storage or depletion rate is zero. Beta angle also has a significant effect because of the influence of heat-rejection performance and water production due to changes in fuel cell idle times.

3.2.4.3 Major Trades and Analyses

The reference PEP thermal control subsystem design is a result of a number of trades and analyses which are summarized in Table 3.2.4-2. A key analysis examined the adequacy of the Orbiter to reject all heat while it was operating in the PEP mode. The analysis showed that PEP operation results in lower heat-rejection requirements, thereby allowing higher electrical power levels than the fuel-cell-powered Orbiter without PEP. Some severe orientations may require flash evaporation cooling or modified orientations for heat-rejection improvements.

Several performance improvements were considered and the option to increase the forward radiator deployment angle from 35 deg to 60 deg was recommended. This option improves rejection from the underside of the panels and reduces environment heat influx for several severe orientations. Change of the deployment angle has a minimal cost impact.

Use of passive cooling for the PEP voltage regulators is a promising option which significantly improves performance for a small cost. The option was studied in the trade called thermal control configuration. The passive concept costs $500,000 more and weighs 38 lb (17.3 kg) more than the reference active design. Benefits include higher allowable power levels of up to 3.4 kW, simplified ground processing, and elimination of Orbiter serial impact for initial installation of 34.5 hr. The serial impact time can correspond to costs as high as $690,000. Safety and reliability are also enhanced by elimination of the fluid interfaces.

3.2.4.4 Thermal Control Subsystem Description

The PEP thermal control subsystem consists of cold plates, lines, and connectors to provide cooling to the PEP voltage regulators. This cooling, amounting to about 3 kW, is required on the sun side of the orbit when power is provided by the solar array.

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Table 3.2.4-2. Summary of Thermal Control Subsystem
Major Trades and Analyses

Trade/analysis	Result
Alternate methods of heat rejection	Separate radiator costly and complex and reduces resources to payloads
	60 deg radiator deployment recommended
	Solid amine CO_2 control not required but will increase capability
Performance analysis	Orbiter adequate for heat rejection
	Flash evaporator system operation required for severe orientations
Thermal control configuration definition, active versus passive	Passive results in: A. Power levels up to 3.4 kW higher B. Adequate temperature control of regulators C. 38 lb more weight and $500,000 higher cost D. Improved safety and reliability E. 34.5 hr serial impact time saved for initial installation F. Simplified ground processing G. Reduced Orbiter SCAR and modication cost
PEP/Orbiter interface analyses	Orbiter aft cold-plate loop can provide PEP cooling
	PEP and Orbiter loop compatible, ΔP, total pressure and temperature
	Physical interface is compatible
EVA considerations	Design criteria established for PEP
	Planned EVA routes may require slight modification
Avionics cooling	Passive cooling adequate
	Heaters required for some components

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The six regulators are mounted on three cold plates, two regulators to each
cold plate. The cold plate has two separate passageways, one for the primary
loop and one for the secondary loop. Freon 21 from the primary aft cold-plate
loop flows through the primary passageways of the three cold plates in paral-
lel. Similarly, the secondary aft cold-plate loop flows through the secondary
cold-plate passageways in parallel. This arrangement maintains separation of
the two loops for safety reasons and balances the heat load equally between
loops.

The physical arrangement is shown in Figure 3.2.4-9. The two aft cold-plate
loops run down either side of the Orbiter bay and disconnects are provided to
interface with the Orbiter loops. Jumpers are installed when the PEP is not
used in the Orbiter. The pressure drop of these jumpers is comparable to the
PEP thermal control loops, thus preventing flow balance change in the Orbiter
system when PEP is not being flown.

PEP avionics components located on the array deployement assembly are not con-
ducive to liquid cooling. They are cooled by passive means with preferential
mounting, thermal coatings, insulation, and electrical heaters. Passive means
are also employed for thermal control of the power-distribution box located on
the PRCA.
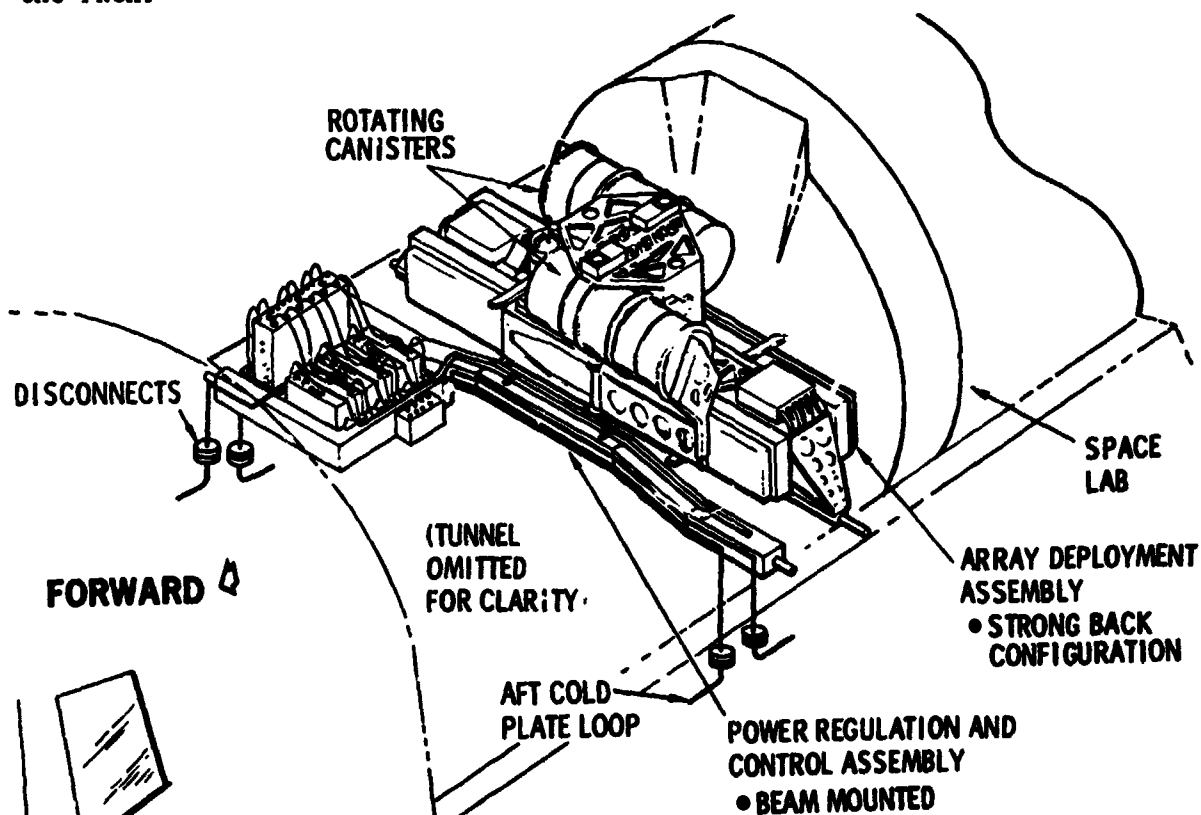
Figure 3.2.4-9. PEP Reference Configuration Mounted in Orbiter

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Section 4
INTERFACE DEFINITION

4.1 PEP EXTERNAL INTERFACES

The PEP, shown in Figure 4-1, contains all the flight hardware which is installed and flown on PEP missions and corsists of the ADA, the PRCA, and an interface kit to accommodate nonstandard interfaces. PEP flight hardware interfaces externally with the Orbiter, the RMS, and the flight crew for operation and monitoring. PEP internal interfaces have been defined between the major hardware elements within the ADA to facilitate procurement of these individual elements.

The PEP hardware, as described above, interfaces directly with the Orbiter and with the RMS. Where standard provisions do not apply, the Orbiter and RMS will accommodate PEP interfaces by means of hardware modifications. These
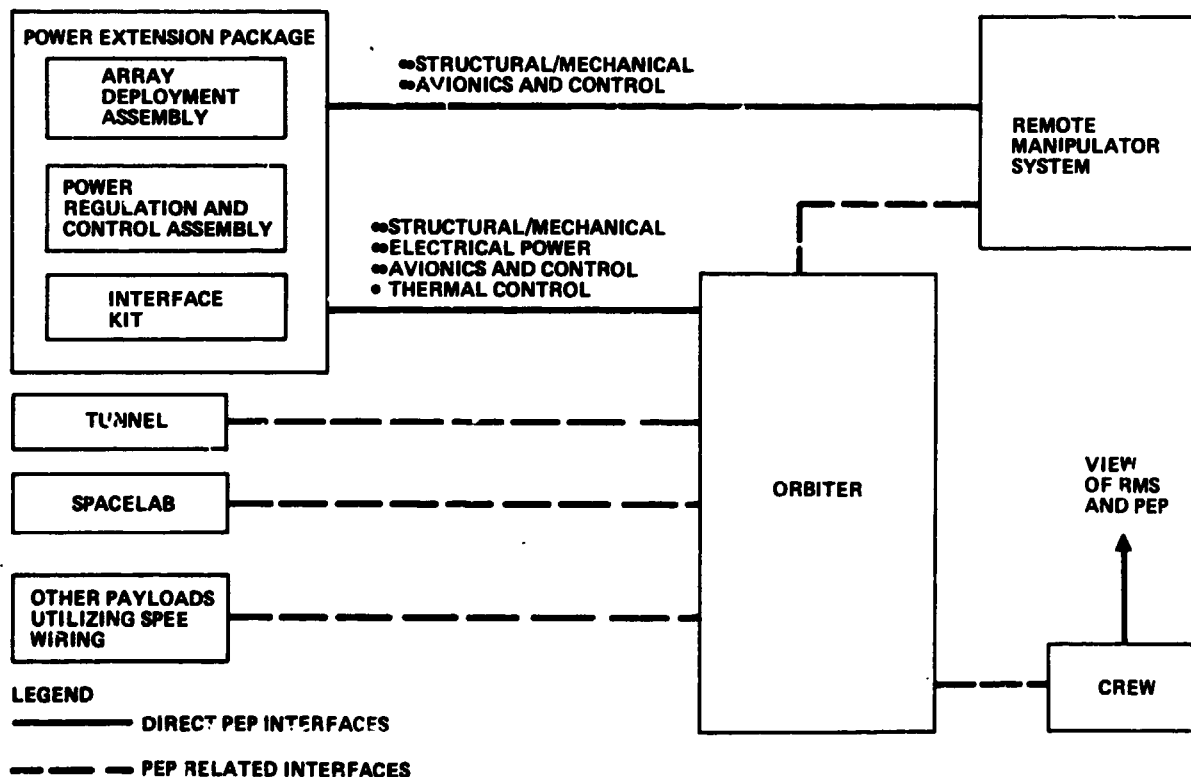
Figure 4-1. PEP Program Elements and Interfaces

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modifications are referred to as scar, since they will generally be flown on non-PEP as well as PEP missions.

The interface between the Orbiter and the RMS is also a consideration for PEP missions for the following reasons:

• The RMS represents the means of communication between the deployable (ADA) and fixed (PRCA) assemblies of the PEP. This communication is further routed via the Orbiter to utilize standard provisions. (For example, in the event the RMS must be jettisoned for safe Orbiter return, wiring is severed by the Orbiter guillotine.)

• The RMS is operated from the AFD crew station for lifting, maneuvering, holding, and returning the PEP ADA. A schematic representation of RMS control and monitoring is shown in Figure 4-2.

HAND CONTROL END EFFECTOR COMMANDED RATES ARE RESOLVED IN GPC TO PROVIDE THE REQUIRED SIX DEGREES OF FREEDOM JOINT RATES

LEGEND
MCIU — MANIPULATOR CONTROLLER INTERFACE UNIT
GPC — GENERAL PURPOSE COMPUTER
RHC — ROTATIONAL HAND CONTROLLER
THC — TRANSLATIONAL HAND CONTROLLER
CRT — CATHODE RAY TUBE
KYBD — KEYBOARD
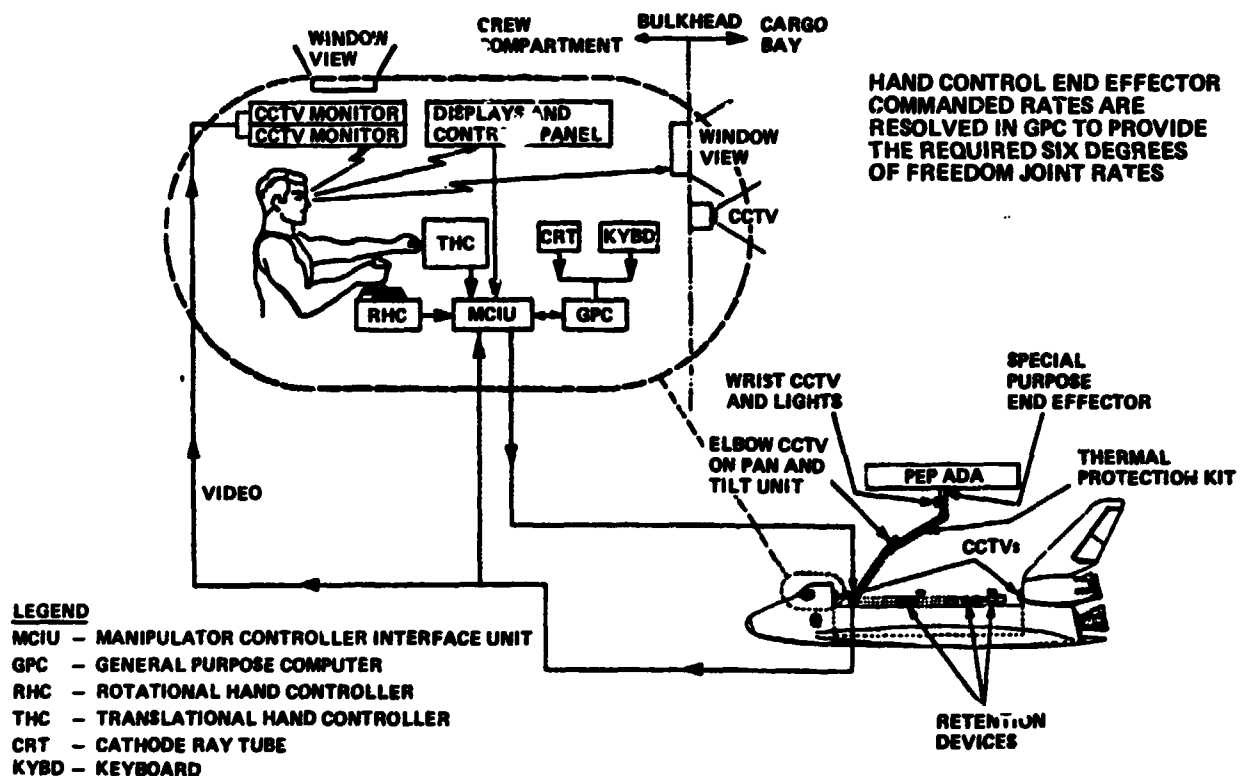
Figure 4-2. RMS Control and Monitoring

In addition to crew operation of the RMS via the Orbiter, crew operation and monitoring of the PEP utilizes display and control provisions at the AFD crew station. Included among these provisions is viewing through windows and via CCTV monitors.

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Interfaces of the Orbiter with the tunnel and with the Spacelab were also considered because of envelope considerations and because these elements share the use of Orbiter bridge fittings with the PEP ADA. In the case of the tunnel, the existing tunnel bridge fittings will not accommodate the PEP ADA and therefore are replaced with Orbiter standard bridge fittings available for payload application.

During operation, the PEP uses the RMS and its SPEE wiring. If another payload on the same mission requires use of the RMS, the PEP ADA will be restowed temporarily in the Orbiter cargo bay to free the RMS for other usage. If this payload also requires use of the RMS SPEE wiring, a means of switching (or sharing) is provided within the Orbiter. Such provision, while not currently standard, is also required to support two or more such payloads on a non-PEP flight.

Figure 4-3 represents an expansion of PEP system interfaces of Figure 4-1. Here, the blocks labeled PEP, RMS, and Orbiter are subdivided into their respective elements that affect interface definition. The subdivision within the RMS and the Orbiter should be considered for reference only; they are identified here only to facilitate understanding of the resulting interfaces.

The following paragraphs will employ Figure 4-3 to summarize the following PEP interfaces:

- PEP to Orbiter Interfaces - Paragraph 4.1.1
- Orbiter to RMS Interfaces - Paragraph 4.1.2
- PEP to RMS Interfaces - Paragraph 4.1.3

The interface designations in parentheses are per Interface Definition Document (IDD), Power Extension Package, SOD 79-0117, which should be consulted for more detailed definition.

4.1.1 PEP to Orbiter Interfaces

Figure 4-3 identifies physical interfaces between PEP hardware elements and the Orbiter. The number prefixes to the interface descriptions below correspond to Figure 4-3 identifiers. From left to right, these interfaces are:

1. Mounting interfaces (S3, S2) between PEP ADA support hardware and Orbiter standard bridge fittings (Figure 4-4). The PEP side of the interface includes three retention latches (two starboard, one port; S3) and a lateral
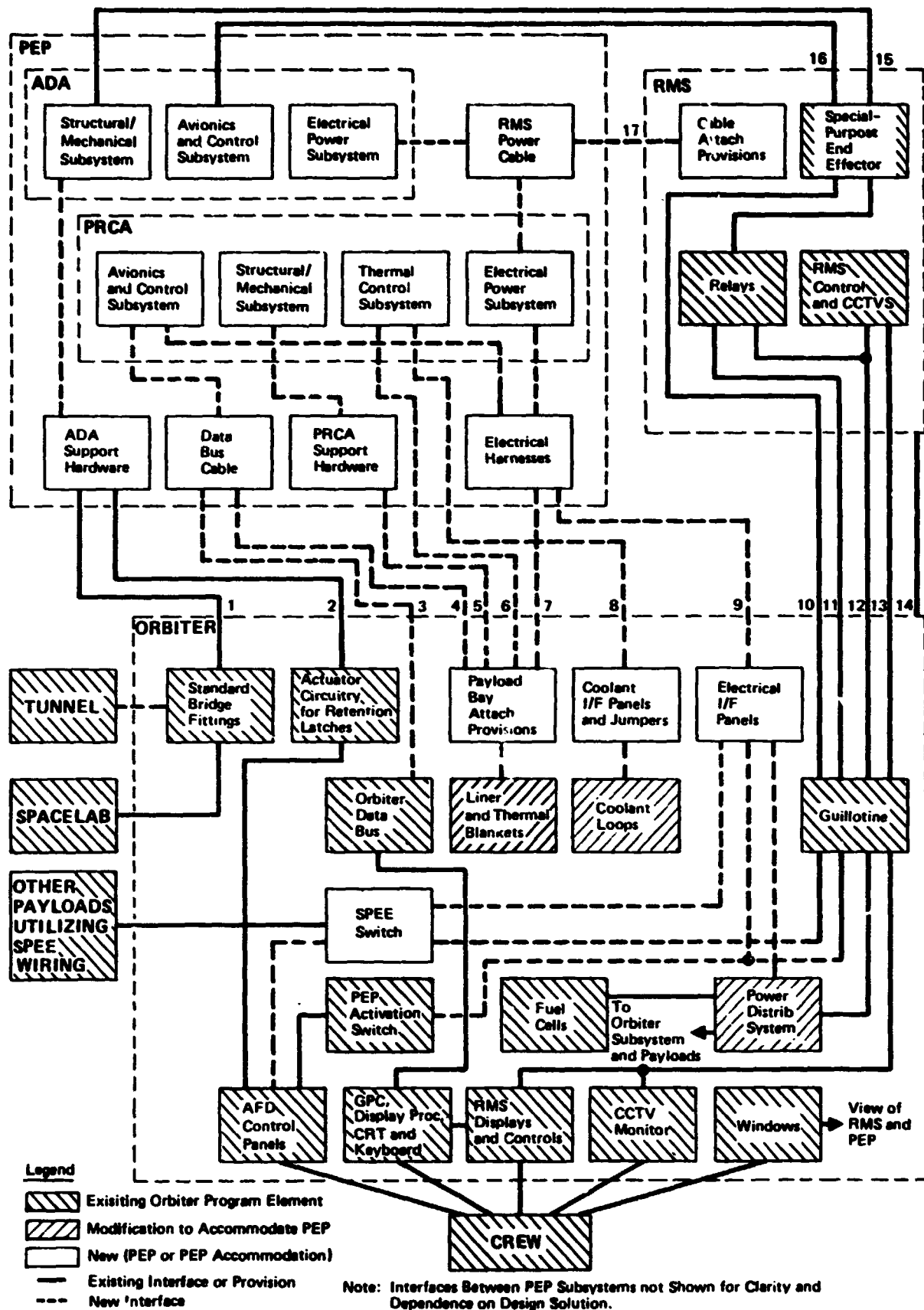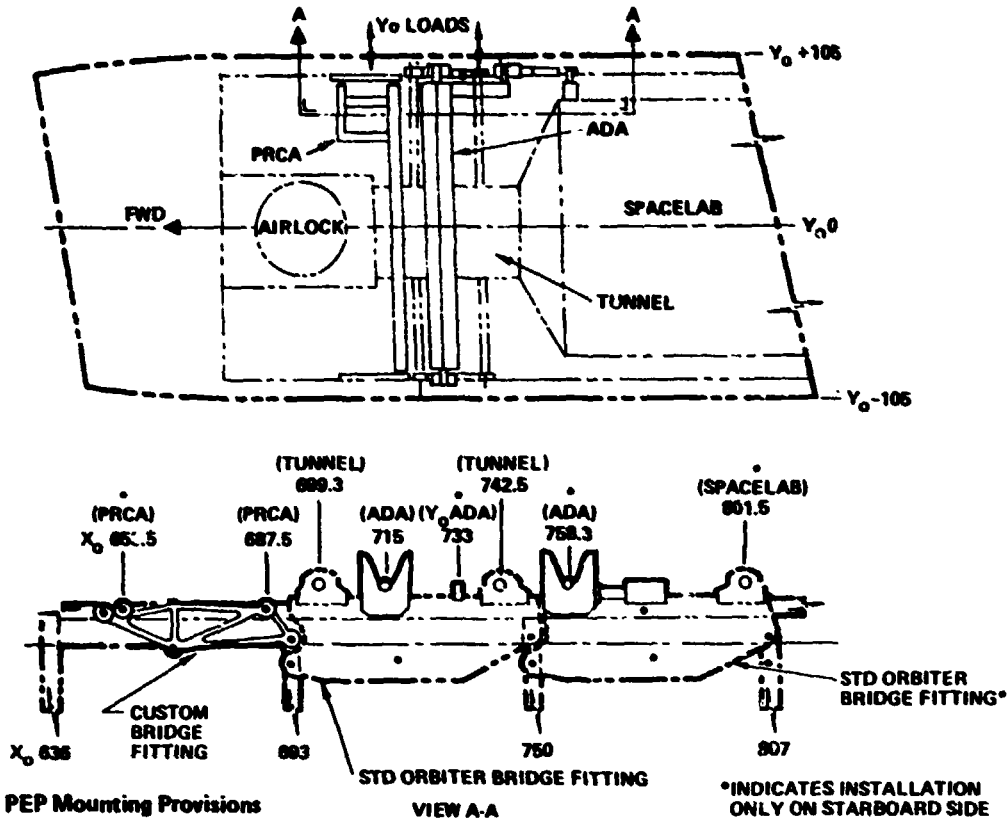
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Figure 4-3. PEP System Interfaces

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Figure 4-4. PEP Mounting Provisions

VIEW A-A

*INDICATES INSTALLATION ONLY ON STARBOARD SIDE

load reaction fitting (starboard; S2). The Orbiter side of the interface includes three bridge fittings (two starboard, one port). The port bridge fitting and the forward starboard bridge fitting are shared by the PEP ADA and the tunnel. The aft starboard bridge fitting is shared by the PEP ADA and the Spacelab. No keel interface exists. Relative deflection is accommodated by the PEP side of the interface.

2. Electrical connector interfaces (P8, D10) at each of above three PEP retention latches for power (P8) and command (D10). The Orbiter side connector halves are standard payload provisions. The crew initiates latch open and close functions at the AFD during ADA deployment and restowage operations.

3. Electrical connector interface (D6) between PEP data bus cable and Orbiter data bus. This interface provides the link between PEP and Orbiter GPC, display processor, CRT and keyboard for crew operation and monitoring of PEP.

4. Cable support interface (S10) between PEP data bus cable and Orbiter clamps provided in payload bay.

5. Mounting interfaces (S4) between PEP PRCA support hardware and Orbiter cargo bay longerons. The PEP side of the interface includes two custom bridge

fittings (one starboard, one port). The Orbiter side of the interface includes bolt holes and bonded spacers (to stand off bridge fittings from side walls). No keel interface exists. Relative deflection is accommodated by PEP side of interface.

6. Plumbing support interface (S6) between PEP PRCA flexible coolant lines and Orbiter clamps provided in payload bay.

7. Electrical harness support interfaces (S8, S14, S16) between PEP electrical harnesses and Orbiter clamps provided in payload bay.

8. Self-sealing Freon 21 disconnect interfaces (F2) between PEP PRCA thermal control susbsystem and two coolant interface panels provided by Orbiter in cargo bay (one starboard, Figure 4-5; one port, Figure 4-6). The PEP side of the interface consists of four disconnect halves on flexible coolant lines that extend from the PRCA. The Orbiter side of the interface consists of four fixed, corresponding disconnect halves. Two disconnects (one supply, one return) are located at each panel. For non-PEP missions, two jumpers are substituted for the PEP lines to provide flow continuity in Orbiter coolant loops. These jumpers are not flown on PEP missions.
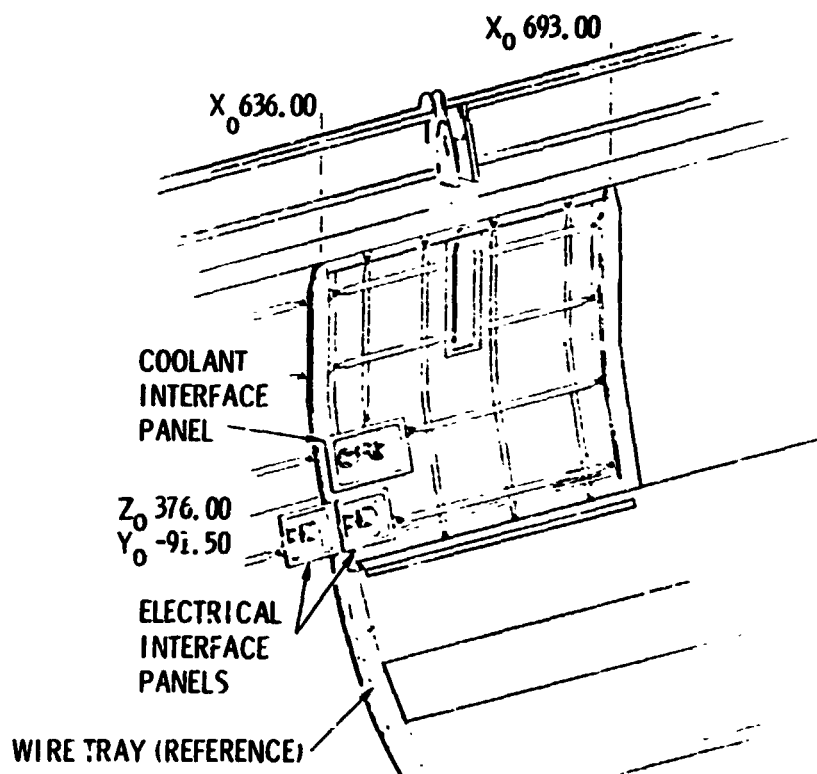

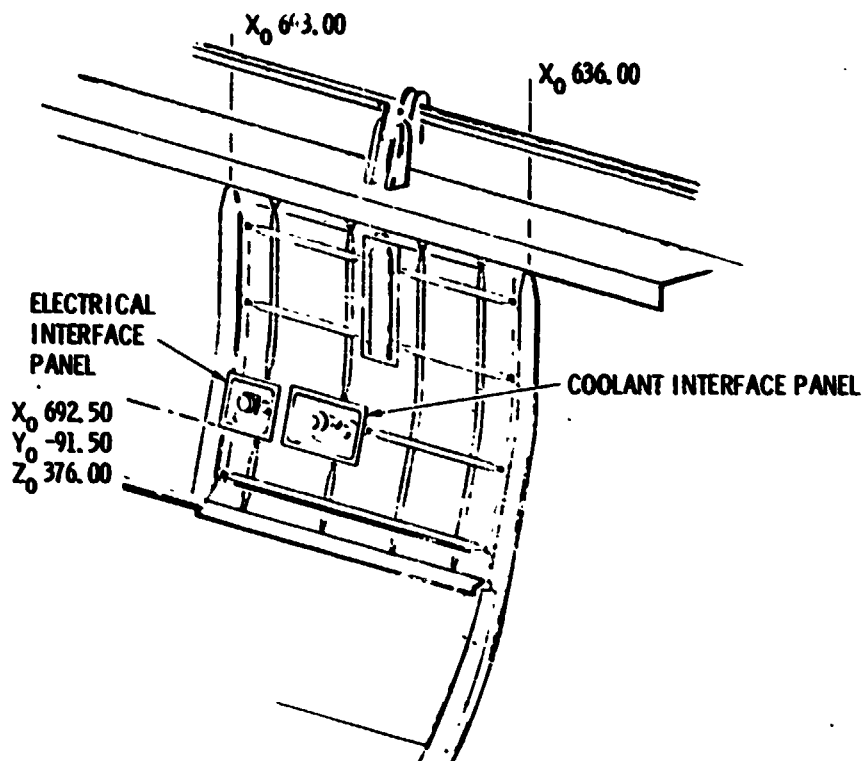Figure 4-5. Orbiter Interface Panels — Starboard

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Figure 4-6. Orbiter Interface Panels — Port

9. Electrical connector interfaces (P4, D2, D4, D7) between PEP electrical harnesses and two electrical interface panels provided by the Orbiter in cargo bay (one starboard, one port).

The starboard panel (Figure 4-5) includes the following connectors:

   a. Regulated power (and returns) from PEP to Orbiter power buses B and C (P4).

   b. Voltage sensing (and returns) by PEP of Orbiter power buses B and C (D2).

   c. Activation signal to enable PEP PRCA to draw current through interface 9a above for initial activation (D4). The crew initiates this signal by a switch provided at the AFD.

   d. Data/command signals that link PEP PRCA with PEP ADA (D7). These signals are routed via Orbiter and RMS in order to utilize existing provisions and to enable sharing of SPEE wiring with another payload.

The port panel (Figure 4-6) includes the following connectors:

   a. Regulated power supply (and returns) from PEP to Orbiter power bus A (P4).

   b. Voltage sensing (and returns) by PEP of Orbiter power bus A (D2).

4.1.2 Orbiter to RMS Interfaces

Figure 4-3 identifies physical interfaces between the Orbiter and the RMS. While these physical interfaces do not affect PEP per se, some of the corresponding functions apply. From left to right, these interfaces are:

10. Electrical connector interface (D12) at RMS shoulder. This is a standard provision, utilized in this case to transmit the data/command signals identified in 9d (Paragraph 4.1.1) from the Orbiter to the RMS.

'1. Electrical connector interface (D12) at RMS shoulder. This is a standard provision utilized in this case to transmit an activation signal to ⊾he RMS relays, that enables PEP ADA to draw current via the interfaces identified in 12 and in 16 (Paragraph 4.1.3) below. The crew initiates this signal by the same AFD switch identified in 9c (Paragraph 4.1.1).

12. Electrical connector interface (P7) at RMS shoulder. This is a standard provision that supplies power for normal RMS functions. In this case, power is also supplied to the RMS relays, identified in 11 above.

13. Electrical connector interface (D9) at RMS shoulder. This is a standard provision that transmits data/command signals for normal RMS functions. There is no PEP interface.

14. Mounting interface (S12) between RMS and Orbiter. This is a standard provision. There is no PEP interface.

4.1.3 PEP to RMS Interfaces

Figure 4-3 identifies physical interfaces between PEP hardware elements and the RMS. These interfaces are:

15. Mechanical mate/demate interface (S13) between the RMS SPEE and the PEP ADA structure. The PEP side of the interface utilizes a standard grapple fixture (Figure 4-7) that is designed to mate with the RMS SPEE. The mate/demate operation is controlled and monitored by the crew from the AFD (Figure 4-2).

16. Electrical connector interface (D8, P6) between the RMS SPEE and the PEP ADA Avionics and Control Subsystem. The PEP side of the interface is a connector half, integral with the grapple fixture identified in Item 15 above. The electrical mating/demating occurs automatically as a result of the mechanical mating/demating. Through this connector are transmitted the data/command signals (D8) identified in 9d (Paragraph 4.1.1) and 10 (Paragraph 4.1.2), together with electrical power (P6) via the RMS relays).
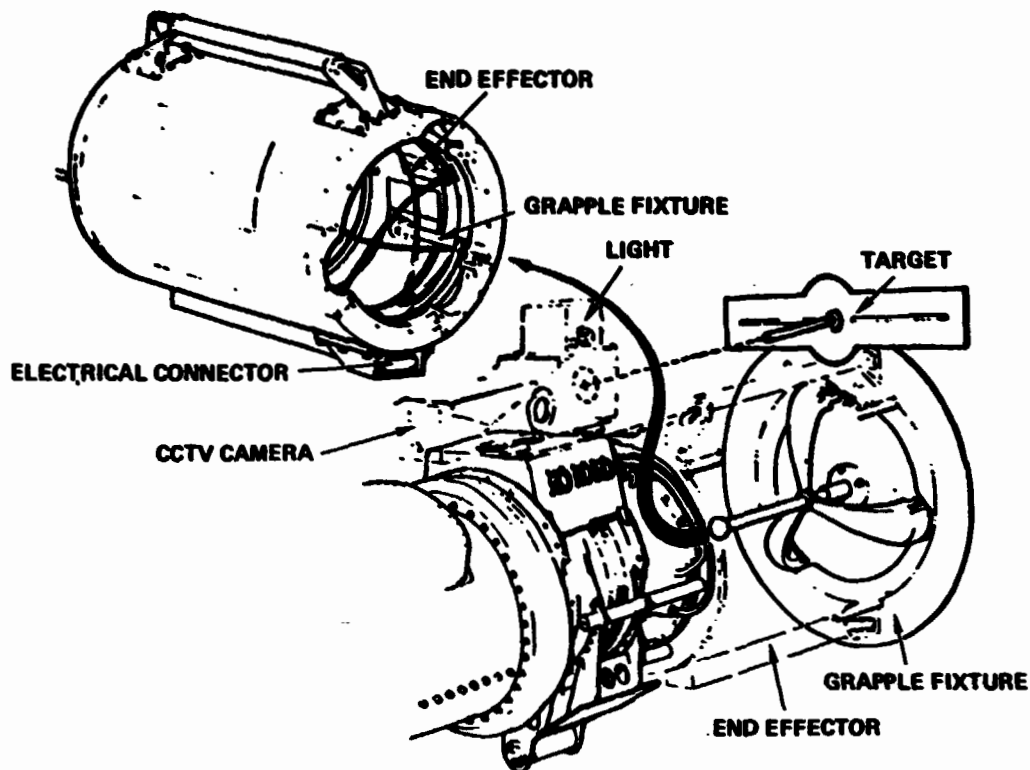
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Figure 4-7. RMS End Effector/Grapple Fixture Interface

17. Cable support interface (S15) between the PEP RMS power cable and the attachment provisions on the RMS. The PEP side of the interface consists of the primary cables that transmit unregulated power from the PEP ADA to the PEP PRCA. The RMS side of the interface consists of clamps and bracketry designed to support the cables in such a manner as to permit RMS joint movement. Mate/demate provisions at both ends of the cables are effected within the PEP side of the interface.

4.2 PEP INTERNAL INTERFACES

Figure 4-8 is a functional block diagram of the PEP reference configuration. During the study, this diagram has been updated to reflect evolution of design and interface definition. The designated interfaces: PEP to Orbiter, Orbiter to RMS, and PEP to RMS are as discussed in Section 4.1.

The interfaces among the hardware elements that comprise the PEP reference configuration are also shown in Figure 4-8. In general, these PEP internal interfaces are controlled within the production drawing system, including procurement specifications.
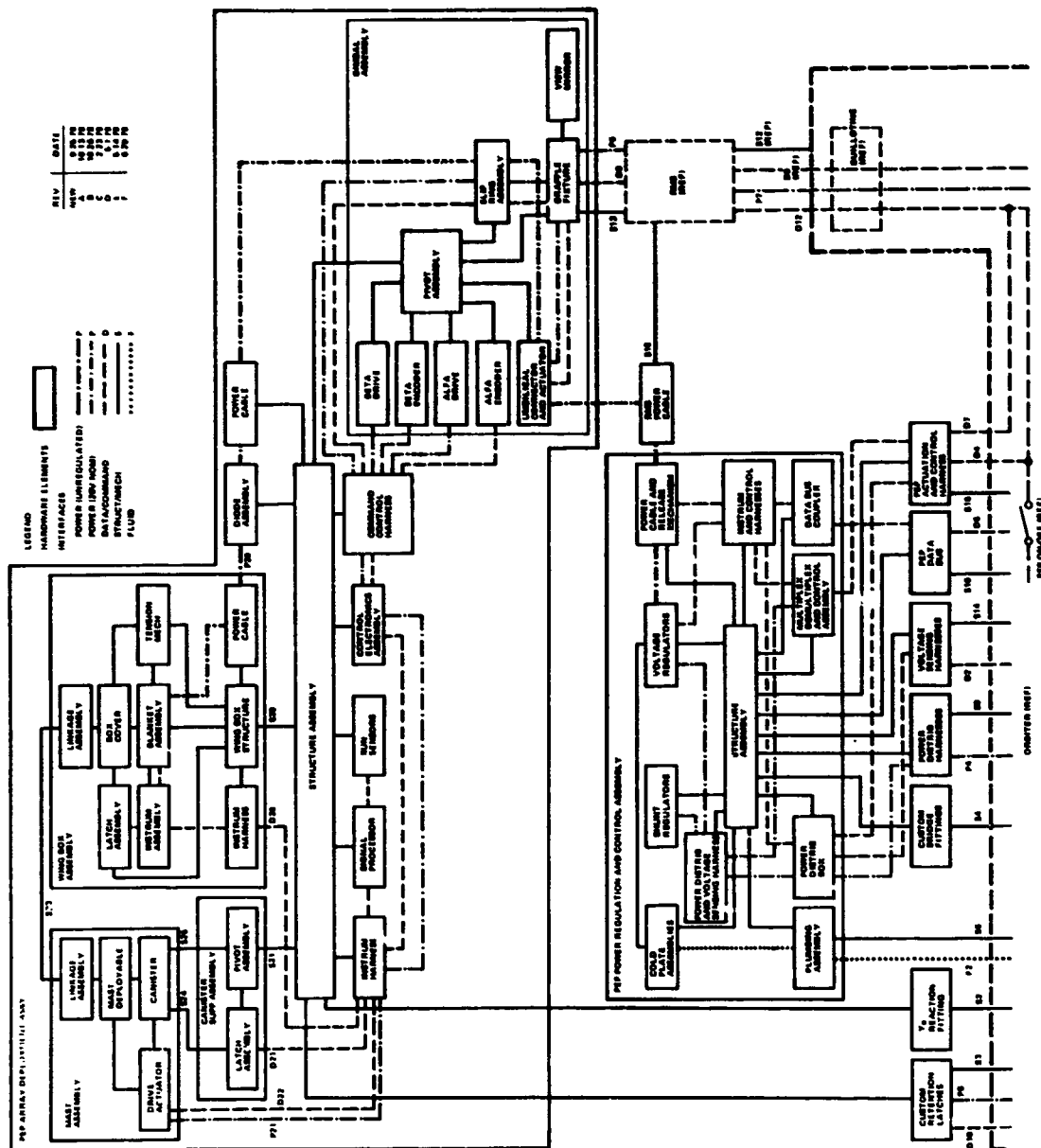
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Figure 4-8. PEP Block Diagram

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Of the procurement items, the solar array represents a key element in terms of interface definition. The following paragraphs treat these interfaces for the PEP reference configuration (Section 4.2.1) and for the solar array module variation of this configuration (Section 4.2.2).

4.2.1 <u>PEP Reference Configuration</u>

The hardware elements that comprise the PEP ADA are illustrated in Figure 4-9. Included among these elements are two wing box assemblies, two mast assemblies, and a single canister support assembly. The interfaces affecting these elements are designated in Figure 4-8 and are tabulated below.

S20   Wing box structural support

P20   Wing box power output

D20   Wing box instrumentation

S23   Wing box/mast mechanical linkage (array deployment)

S25   Canister structural support (rotation, suspension, and lockout)

S24   Canister latching (stowage)

P21   Mast drive actuator power
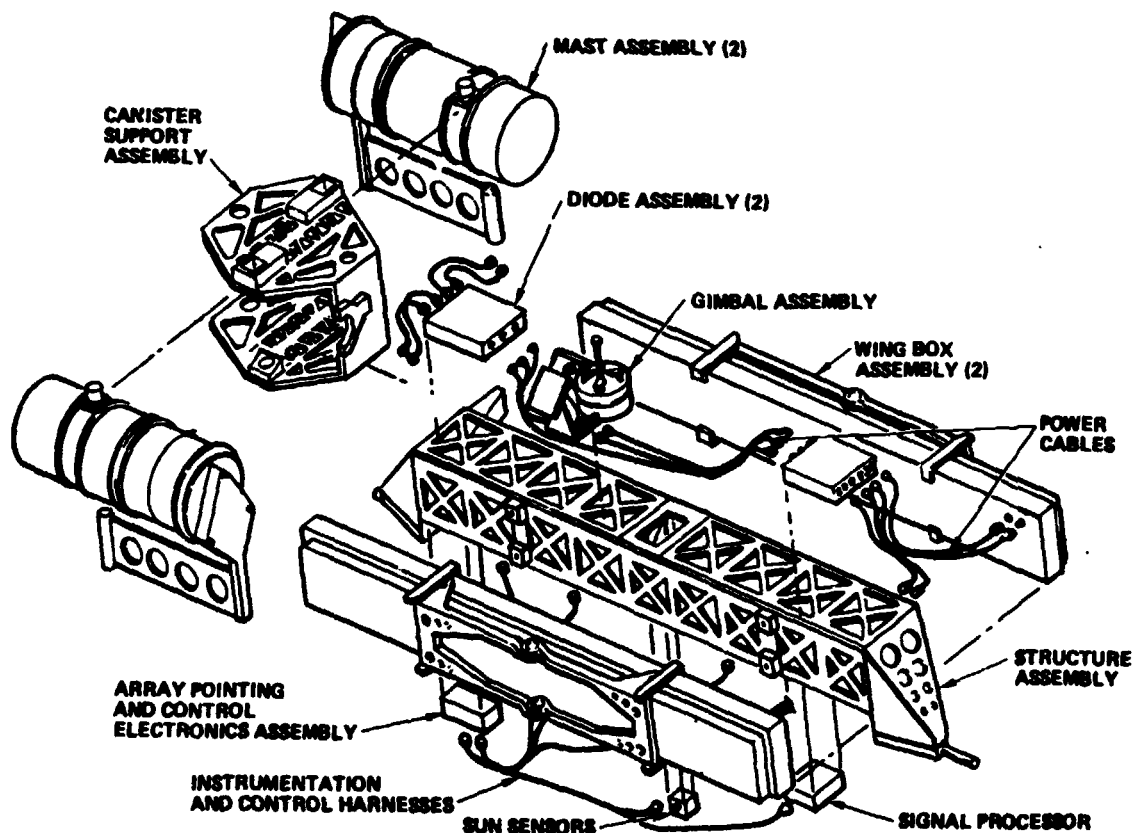
D22   Mast assembly instrumentation and control

Figure 4-9. PEP ADA — Reference Configuration

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D21 Latch assembly instrumentation and control

S21 Pivot assembly structural support

If more than one of the hardware elements are procured from the same supplier, then the interfaces between these elements are controlled within that supplier's drawing system (e.g., Interface S23, wing box/mast mechanical linkage would be the internal responsibility of a supplier of both the wing box assembly and the mast assembly). Interfaces between supplier hardware elements and other elements are controlled by procurement specification.

4.2.2 Solar Array Module Variation

A variation to the PEP ADA reference configuration exists that includes two solar array modules (Figure 4-10). Each module incorporates the hardware necessary to perform the combined functions of the wing box assembly, mast assembly, and canister support assembly of the reference configuration. Referring to Figure 4-8, and the tabulation of Section 4.2.1, Interface S21 then becomes an internal solar array module interface (i.e., mounting is directly to the wing box structure). Thus, Interface S21 and internal Interfaces S23, S24, and S25 are controlled within the solar array module supplier's drawing system.

The remaining external Interfaces P20, S20, D20, D21, D22, and P21 are controlled by the solar array module procurement specification.
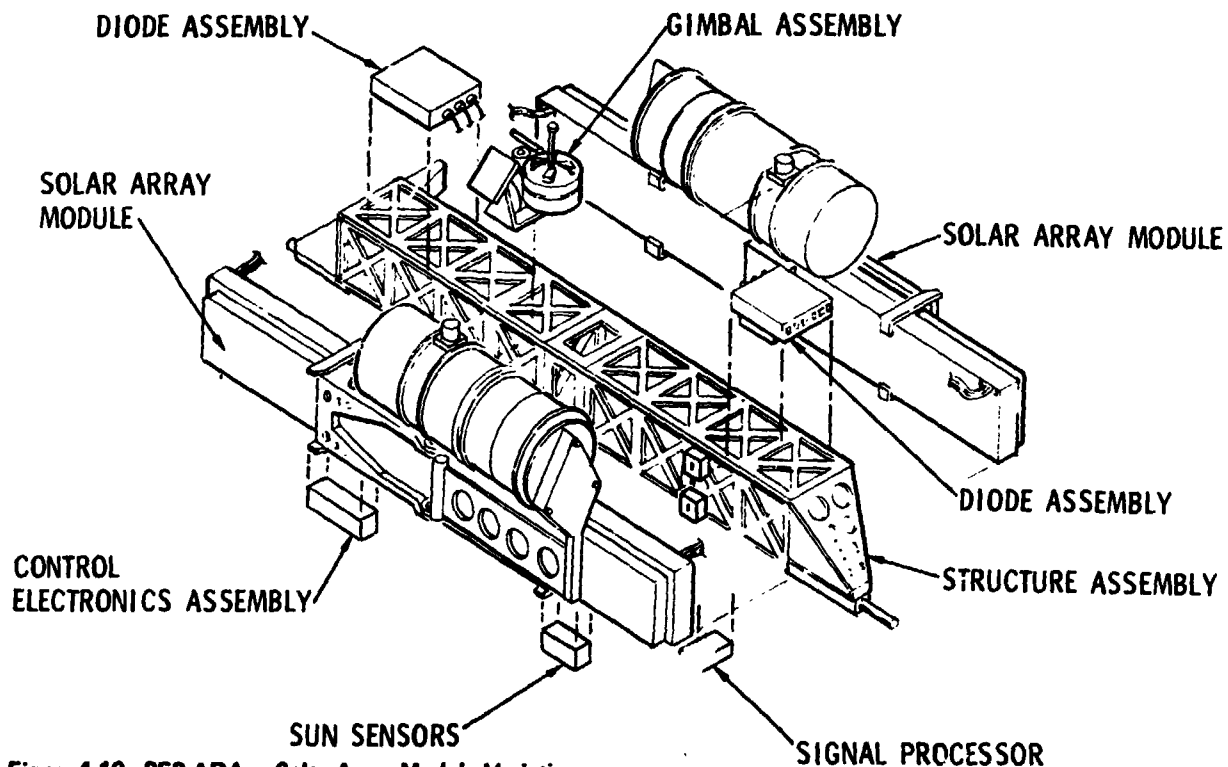

Figure 4-10. PEP ADA — Solar Array Module Variation

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